VOLUME I
AIRCRAFT PERFORMANCE

CHAPTER 12
DATA REDUCTION AND CORRECTIONS TO STANDARD DAY

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12.1 INTRODUCTION

An ancient report, from the early days of flying, discussing the crash of a "Jenny" shortly after takeoff on a hot summer day, concluded that the primary cause was "there was no lift in the air that day." The determination of the effects of nonstandard atmospheric conditions on aircraft performance has come a long way since then. These determinations are particularly important in flight test, since performance specifications must be written for some set of "standard" conditions, and flight tests are not usually conducted on "standard" days. Modern computer data reduction capabilities have greatly reduced the manual labor required for performance calculations. On the other hand, they tend to hide assumptions and factors which can turn out to be extremely important in performance testing.

This section examines how test day performance data may be reduced or related to performance under standard conditions. Emphasis is placed on the relationship between data reduction and available instrumentation. The techniques are those used in the TPS performance data reduction programs, which are similar to those used in AFFTC flight test programs.

12.2 STANDARD CONDITIONS

The procuring activity, usually the system program office (SPO), normally determines which conditions will be considered standard for performance specification compliance. Some of the parameters which must be considered are discussed below.

12.2.1 Atmospheric Conditions

These include the variation of ambient temperature and static pressure with altitude. Depending on the aircraft mission, a standard, Arctic standard, tropical standard, or some combination will be specified. Standard day conditions are discussed in Chapter 5. The TPS data reduction programs use the 1962 U.S. Standard Atmosphere.

12.2.2 Weight

A standard weight must be specified since weight affects the angle of attack, drag, and acceleration characteristics for a given set of flight conditions. This depends on the aircraft mission and several standard weights will normally be specified for various portions of the mission profile. To determine standard weight for any aircraft:

a. The average maximum gross weight for the aircraft is found in the Flight Manual or from manufacturer's data.

b. The average fuel used during engine start and ground maneuvering is subtracted from the maximum gross weight to determine the standard weight for takeoff data reduction.

c. The aircraft weight at level off at a particular altitude is the standard weight for level accels, sawtooth climbs and check climbs.

   1. Compute the sea level standard weight by subtracting the fuel required to accelerate to climb speed from the takeoff standard weight.

   2. Subtract the fuel required to climb to different altitudes from the standard weight at sea level. This gives a standard weight at each altitude.

d. Determine the MIL-C-5011A fuel reserve. This weight is added to the aircraft empty weight and used as a standard weight for descent performance. (e.g. 5% of initial fuel + 20 minutes at sea level at maximum endurance.)

e. The standard weights for performance tests such as turns, W/δ planning, or ferry range missions can be determined by averaging the results from c and d.

12.2.3 Center of Gravity (CG)

The CG position determines the elevator (or slab) position required to maintain a given set of flight conditions. This directly affects parasite drag and can have a large effect on performance. (The TPS programs do not correct for CG position.)

12.2.4 Wind

No-wind conditions are specified in most cases. Wind and wind shear influence climb and descent performance. (The TPS programs do not correct for wind.)

12.2.5 Configuration

Standard configurations must be defined and tested. In TPS performance testing, only the cruise (clean) and speed-brake out (for penetration descents) configurations are specified.

12.2.6 Schedules and Techniques

Climb, descent, cruise, and acceleration schedules must be specified for specific performance determinations. Off-schedule tests may be corrected to standard schedules but the TPS programs do not include these corrections.

12.2.7 Other Considerations

Other factors may be specified depending on the aircraft. These can include the number of operating engines, trim position, and several other factors. Specific requirements must be stated for each flight condition and desired performance parameter.

12.3 PITOT-STATIC DATA REDUCTION

Altitude and airspeed are the two variables that are of primary concern throughout performance and flying qualities testing. Unfortunately, the measurement of these variables by the aircraft's pitot-static system is complicated by the errors discussed in Chapter 5.

12.3.1 Tower Fly-by Data Reduction

Position error is found from tower fly-by data using the following:

1. Find the test aircraft H_c.

   a. Plot pressure altitude versus time as shown in Figure 12.1 from the preflight and postflight ground blocks. This is called the ground block method. Ramp H_c is the pressure altitude read on the altimeter corrected for instrument error.

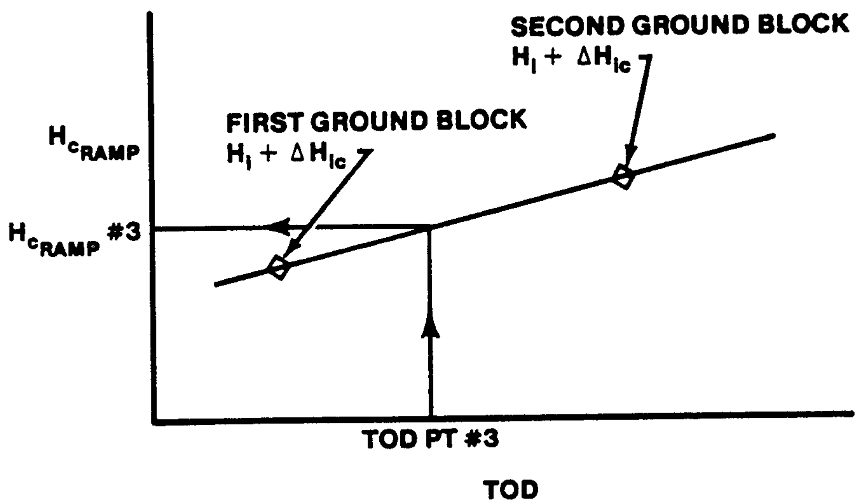

FIGURE 12.1. GROUND BLOCK PRESSURE ALTITUDE PLOT

$$H_{c_{Ramp}} = H_{i_{Ramp}} + \Delta H_{ic} \qquad (12.1)$$

This plot provides ramp H_c for anytime during the flight.

   b. Find theodolite H_c by adding the difference in elevation between the ramp and the theodolite.

$$H_{c_{Theodolite}} = H_{c_{Ramp}} + \left(\text{Elevation}_{\text{Theodolite}} - \text{Elevation}_{\text{Ramp}} \right) \frac{T_{as}}{T_a} \qquad (12.2)$$

An alternate method to obtain $H_{c_{Theodolite}}$ is to read it directly in a theodolite tower with a sensitive pressure instrument at the tower zero gridline.

c. Find the test aircraft H_c by adding theodolite reading, TR, and correcting for nonstandard temperature.

$$H_{c_{\text{Test Aircraft}}} = H_{c_{\text{Theodolite}}} + \frac{T_{as}}{T_a} \text{ TR } 31.4 \qquad (12.3)$$

where 31.4 is the geometric conversion factor for the Edwards Tower - Theodolite System.

2. Find instrument-corrected altitude H_{ic}, velocity V_{ic}, static pressure ratio $P_s/P_{a_{sl}}$, differential pressure ratio $q_{cic}/P_{a_{sl}}$, and instrument-corrected Mach M_{ic}.

a. $$H_{ic_t} = H_i + \Delta H_{ic} \qquad (12.4)$$

b. $$V_{ic_t} = V_i + \Delta V_{ic} \qquad (12.5)$$

c. Use the low altitude altimeter equation for:
t SL

$$P_{s_t}/P_{a_{sl}} = \delta_{ic_t} = \left[1 - 6.87559 \times 10^{-6} H_{ic_t}\right]^{5.2559} \qquad (12.6)$$

d. Use the subsonic calibrated airspeed equation for:

$$q_{cic_t}/P_{a_{sl}} = \left[1 + 0.2\left(\frac{V_{ic_t}}{a_{sl}}\right)^2\right]^{3.5} -1 \qquad (12.7)$$

e. Use the Mach equation for:

$$M_{ic} = \sqrt{5\left[\left(\frac{q_{cic_t}}{P_{s_t}} + 1\right)^{2/7} - 1\right]} \qquad (12.8)$$

where

$$\frac{q_{cic_t}}{P_{s_t}} = \frac{\dfrac{q_{cic_t}}{P_{a_{sl}}}}{\delta_{ic_t}} \qquad (12.9)$$

3. Standardize ΔH_{pc} and V_{ic}. Altimeter position error and instrument corrected velocity are standardized to 2,300 feet for the tower fly-by test at Edwards.

a. Find ΔH_{pc_t}. The necessary information is available to compute altimeter position error correction ΔH_{pc} at the test H_{ic} and M_{ic}. The test points were taken in steady state conditions when lag correction is zero so the test altimeter position error correction is computed according to

$$\Delta H_{pc_t} = H_{c_t} - H_{ic_t} \qquad (12.10)$$

This is one form of position error for the test conditions H_{ic}, M_{ic}, and gross weight.

b. Correct ΔH_{pc_t} to $\Delta H_{pc_{2300}}$ (12.11)

The altimeter position error must be standardized to one altitude. Each data point was taken at a different instrument corrected altitude, H_{ic}, and must be corrected to a common H_{ic} for comparison. When altitude changes are small or angle of attack effects are not significant the following correction can be made:

$$\Delta H_{pc_{2300}} = \Delta H_{pc_t} \frac{\theta_{s_{2300}}}{\theta_{s_t}} \qquad (12.12)$$

where θ_{s_t} is the standard temperature ratio evaluated at the H_{ic} of the test aircraft.

$$\theta_{s_t} = 1 - 6.87559 \times 10^{-6} H_{ic_t} \qquad (12.13)$$

c. Correct V_{ic_t} to $V_{ic_{2300}}$. Altimeter position error is assumed to be a function of M_{ic} and H_{ic} rather than V_{ic} and H_{ic} so a corrected V_{ic} is computed assuming the data point was flown at an H_{ic} of 2,300 feet and the test M_{ic}. From Equation 12.8, the differential pressure ratio for the standard altitude of 2,300 feet is the same as that for the test altitude.

$$\frac{q_{cic_{2300}}}{P_{s_{2300}}} = \frac{q_{cic_t}}{P_{s_t}} \qquad (12.14)$$

From

$$\frac{q_{cic_{2300}}}{P_{a_{sl}}} = \left(\frac{q_{cic_t}}{P_{s_t}}\right) \delta_{2300} \qquad (12.14)$$

$V_{ic_{2300}}$ can be computed using the calibrated airspeed equation.

$$V_{ic_{2300}} = a_{sl}\sqrt{5\left[\left(\frac{q_{cic_{2300}}}{P_{a_{sl}}} + 1\right)^{2/7}\right] - 1} \qquad (12.16)$$

4. The parameter nW/δ_{ic} is necessary to evaluate the angle of attack effect. First, determine the most representative gross weight. The gross weight and speed determine what angle of attack is required for flight at a given Mach during a level steady state test point. The gross weight during the low speed test points is most representative because of the relatively high angle of attack.

$$nW/\delta_{2300} = \frac{\text{Average Low Speed Gross Weight}}{\delta_{2300}} \qquad (12.17)$$

The ratio W/δ cannot be held constant for a tower fly-by test so all low speed points should be flown at nearly the same gross weight. During pacer missions, W/δ may be held constant by increasing altitude as the gross weight decreases to eliminate the angle of attack effect entirely.

Sideslip angle may also affect position error. If sideslip angle effects are suspected, data points should be repeated with known variations in sideslip angle.

5. Calculate the other forms of position error, position error pressure ratio $\Delta P_p/P_s$, airspeed position error correction ΔV_{pc}, Mach position error correction ΔM_{pc}, and position error pressure coefficient, $\Delta P_p/q_{cic}$.

a. $$\Delta P_p/P_s = \frac{3.61382 \times 10^{-5}}{\theta_{s_t}} \Delta H_{pc} \qquad (12.18)$$

where θ_{s_t} is the standard atmospheric temperature ratio

$$\theta_{s_t} = 1 - 6.87559 \times 10^{-6}\ H_{ic} \qquad (12.19)$$

Position error pressure ratio is the difference between static and ambient pressure to ambient pressure

$$\frac{\Delta P_p}{P_s} = 1 - \frac{\delta}{\delta_{ic}} \qquad (12.20)$$

and is a convenient form of position error for determining ΔV_{pc} and ΔM_{pc}.

b. $$\Delta V_{pc} = \frac{a_{sl}^2\ \delta_{ic}}{1.4\ V_{ic}\left[1 + 0.2\left(\frac{V_{ic}}{a_{sl}}\right)^2\right]^{2.5}} \frac{\Delta P_p}{P_s} \qquad (12.21)$$

This conversion is good for $V_{ic} \leq 661.48$ knots.

c. $$\Delta M_{pc} = \frac{\left(1 + 0.2\, M_{ic}^{2}\right)}{1.4\, M_{ic}} \frac{\Delta P_p}{P_s} \qquad (12.22)$$

This conversion is good for $M_{ic} \leq 1.0$.

d. $$\frac{\Delta P_p}{q_{cic}} = \frac{\Delta P_p/P_s}{q_{cic}/P_s} \qquad (12.23)$$

This position error pressure coefficient should be plotted as a function of M_{ic}.

12.3.2 Pacer Data Reduction

With the exception of finding H_c and V_c, the data reduction is the same as for tower fly-by.

1. Find pacer aircraft H_c and V_c

$$H_{c_p} = H_{i_p} + \Delta H_{ic_p} + \Delta H_{pc_p} \qquad (12.24)$$

$$V_{c_p} = V_{i_p} + \Delta V_{ic_p} + \Delta V_{pc_p} \qquad (12.25)$$

Equations are available that represent position error for the pacer aircraft.

2. Find the test aircraft H_{ic_t} and V_{ic_t} as in the tower fly-by data reduction.

3. Determine altimeter position error ΔH_{pc} and airspeed position error ΔV_{pc}.

   a. Assume $H_{c_{\text{test aircraft}}} = H_{c_{\text{pacer aircraft}}}$ and $V_{c_{\text{test aircraft}}} = V_{c_{\text{pacer aircraft}}}$ since aircraft were level and co-speed.

   b. Find ΔH_{pc_t} and correct it to the desired test altitude as in tower fly-by data reduction.

c. $\Delta V_{pc_t} = V_c - V_{ic_t}$ (12.26)

This yields airspeed position error ΔV_{pc} independently from the altitude method and provides a comparison to see if total pressure P_T is fully recovered.

d. Find ΔH_{pc_t} from the ΔV_{pc_t} using Equations 12.18 and 12.21.

e. Correct ΔH_{pc_t} determined from ΔV_{pc_t} to the desired test altitude and compare the results.

12.3.3 Radar Data Reduction

1. Select data point acquired from on-board instrumentation and correct for instrument error, if necessary.

$$H_{ic} = H_i + \Delta H_{ic} \tag{12.27}$$

2. Determine

$$H_{c_{pace}} = H_{i_p} + \Delta H_{ic_p} + \Delta H_{pc_p} \tag{12.28}$$

3. Change tapeline difference to pressure altitude difference through the temperature correction

$$\Delta H_c = \Delta h \frac{T_{a_s}}{T_{a_t}} \tag{12.29}$$

where $\Delta h = H_{R_{test}} - H_{R_{pace}}$ and H_R represent radar measured altitude.

4. Determine test pressure altitude and position error and standardize to the test altitude

$$\Delta H_{pc_t} = H_{c_t} - H_{ic} \tag{12.30}$$

5. Calculate velocity position error from ΔH_{pc} as in the tower fly-by method.

12.3.4 Speed Course Data Reduction

1. Find the average true airspeed V_T.

$$V_T = 1800 \left(\frac{D}{t_1} + \frac{D}{t_2}\right) \qquad (12.31)$$

where D is the course length in nautical miles and t_1 and t_2 are the course times in seconds.

2. Ambient temperature: $T_a = (T_i + \Delta T_{ic} + 273.16) - \frac{K_t V_T{}^2}{7592}$ (12.32)

where K_t is the temperature recovery factor.

3. $$M = \frac{V_T}{38.967 \sqrt{T_a}} \qquad (12.33)$$

4. $H_{ic_t} = H_i + \Delta H_{ic}$

$V_{ic_t} = V_i + \Delta V_{ic}$

5. a. $$\frac{q_{cic_t}}{P_{a_{sl}}} = \left[1 + 0.2\left(\frac{V_{cic_t}}{a_{sl}}\right)^2\right]^{3.5} - 1 \qquad (12.34)$$

b. $$\delta_{ic_t} = \left[1 = 6.87559 \times 10^{-6} H_{ic_t}\right]^{5.2559} \qquad (12.35)$$

c. $$M_{ic} = \sqrt{5\left[\left(\frac{q_{cic_t}}{P_{s_t}} + 1\right)^{2/7} - 1\right]} \qquad (12.36)$$

where

$$\frac{q_{cic_t}}{P_{s_t}} = \frac{\dfrac{q_{cic_t}}{P_{a_{sl}}}}{\delta_{ic_t}} \qquad (12.37)$$

6. $\Delta M_{pc} = M - M_{ic}$

   Mach position error correction does not need to be standardized. It always remains constant over small altitude changes and remains constant over large altitude changes if angle of attack effects are not significant.

7. Find position error pressure ratio $\Delta P_p/P_s$, altitude position error correction ΔH_{pc}, and airspeed position error correction ΔV_{pc} as in Section 5.7.2.3.

$$\frac{\Delta P_p}{P_s} = 1.4 \left[\frac{M_{ic} + \dfrac{\Delta M_{pc}}{2}}{1 + 0.2 \left(M_{ic} + \dfrac{\Delta M_{pc}}{2} \right)^2} \Delta M_{pc} \right] \qquad (12.38)$$

8. Correct V_{ic_t} to $V_{ic_{2300}}$

Standardize V_{ic} to the instrument correct airspeed that would have been computed if the test Mach had been flown at the standard test altitude.

12.4 TAKEOFF DATA REDUCTION

Takeoff data will be recorded on all missions during the performance phase. The purpose is to determine the ground roll at the instant of lift-off and to correct this data for nonstandard conditions. While cinetheodolite data is the most accurate, the resources required prohibit its acquisition on all missions. Most data must come from pilot estimates. To improve the quality of this pilot estimated data, there are certain steps that should be taken.

1. Obtain current pressure altitude and temperature from ATIS or ground control just before takeoff; also record fuel used or remaining.

2. Record the wind speed and direction given by the tower with takeoff clearance.

3. Align the aircraft with a runway light for run-up. Consider selecting the appropriate light so predicted liftoff will be abeam a runway marker; e.g., with a predicted 2600 feet roll, line up three lights before a runway marker and expect to be airborne after passing two additional runway markers.

4. Assign one crewmember primary responsibility for obtaining takeoff airspeed, the other crewmember for distance, but attempt to verify each other's readings.

5. Take off using the test team's standardized procedures as specified in the test plan.

Reduction of takeoff data to standard conditions is not difficult but becomes rather tedious when accomplished manually due to the large number of data points required. By using Equations 12.41, 12.42 and 12.43 as a basis to correct for wind, slope, thrust, weight, and density, the test day data can be corrected for nonstandard conditions.

The use of these equations requires test day ground roll, ground speed, headwind, runway slope, aircraft test and standard weight, test and standard day air density and thrust. To determine these, collect the following data:

1. Takeoff speed (kts, V_{ic} for pilot data, V_{GS} for Cinetheodolite)

2. Ground roll distance, S_{g_t} (ft)

3. Fuel remaining (lbs)

4. Runway temperature (^{o}C)

5. Pressure altitude (ft)

6. Wind speed, V_w (kts)

7. Wind direction, X_w (deg magnetic)

8. Runway heading, X_r (deg magnetic)

9. Runway slope angle, θ (deg)

Ground speed at liftoff is obtained directly from Cinetheodolite data. For pilot estimated data, indicated airspeed at liftoff must be used along with pressure and temperature to calculate true airspeed. Subtracting headwind yields the ground speed.

a. Calculate headwind

$$V_{w_h} = V_w \cos (X_w - X_r) \tag{12.39}$$

b. Calculate true airspeed and ground speed. For cinetheodolite data, ground speed is measured and true airspeed is calculated.

$$V_T = V_{GS} + V_{w_h} \tag{12.40}$$

For crew observed data, true airspeed, V_T, is obtained using pitot-static equations. Ground speed is calculated by subtracting the test day headwind.

$$V_{GS} = V_T - V_{w_h} \tag{12.40a}$$

c. Correct ground roll for wind

$$S_{g_{cw}} = S_{g_t} \left(\frac{V_T}{V_{GS}} \right)^{1.85} \tag{12.41}$$

d. Correct ground roll for slope

$$S_{g_{cs}} = \frac{S_{g_{cw}}}{1 - \dfrac{2g\, S_{g_{cw}} \sin \theta}{V_{GS}^2}} \tag{12.42}$$

e. Standardize ground roll to sea level standard day for the test day true airspeed by applying thrust, weight, and air density, corrections.

$$S_{g_s} = S_{g_{cs}} \left(\frac{W_s}{W_t}\right)^{2.3} \left(\sigma_t\right)\left(\frac{F_{n_t}}{F_{n_s}}\right)^{1.3} \qquad (12.43)$$

The terms F_{n_t} and F_{n_s} represent the test day and standard day net thrust respectively. These values are calculated from the model furnished by the manufacturer in the engine thrust deck. For example, Appendix D contains simplified thrust curves for the A-37B, T-38A and RF-4C aircraft. Knowledge of θ, δ, and M is sufficient to determine net thrust from these curves.

The weight and air density corrections to ground roll include not only their effect on acceleration, but also their effect on correct takeoff speed. A similar correction must be applied to the takeoff speed.

f. Standardize takeoff speed for weight and air density.

$$V_{T_s} = V_T\left[\left(\frac{W_s}{W_t}\right)\sigma_T\right]^{1/2} \qquad (12.44)$$

Standardized ground roll should be plotted as a function of standardized takeoff speed as shown in Figure 12.2.

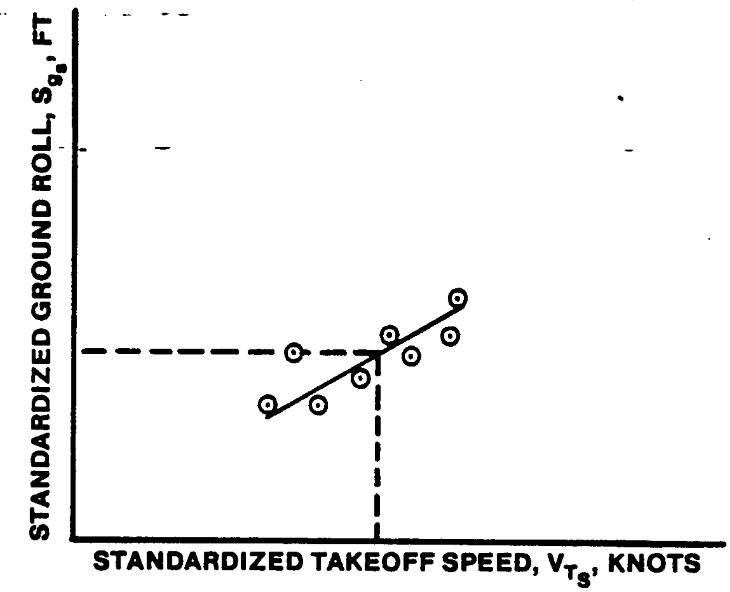

FIGURE 12.2. GROUND-ROLL TAKEOFF DISTANCE

There will be significant scatter in the data points since it is difficult to determine liftoff point and the corresponding ground-roll. Use of cinetheodolite data greatly improves the data, although most data will still be crew estimated. Since some data will represent higher than nominal takeoff speeds or late readings, and some will represent lower than nominal speeds or early readings, a curve through the data should be roughly parabolic. This corresponds to the curve one would obtain during a ground acceleration test. By plotting the final portion of the takeoff acceleration obtained from cinetheodolite missions, the shape of the curve can be obtained. This will assist in drawing a curve through the large number of data points. The sea level standard day takeoff distance is then read from the curve at the accepted takeoff speed.

12.5 ENERGY METHOD DATA REDUCTIONS

Energy characteristics are of paramount importance to the effectiveness of fighter aircraft. Specific excess power P_s, as a function of Mach, and altitude H, graphically display an aircraft's ability to climb, accelerate and turn.

12.5.1 Excess Thrust

Every performance parameter, except fuel flow, is a direct function of excess thrust

$$F_{ex} = F_n - D \tag{12.45}$$

where F_n = net thrust (along the flight path) and D = total drag

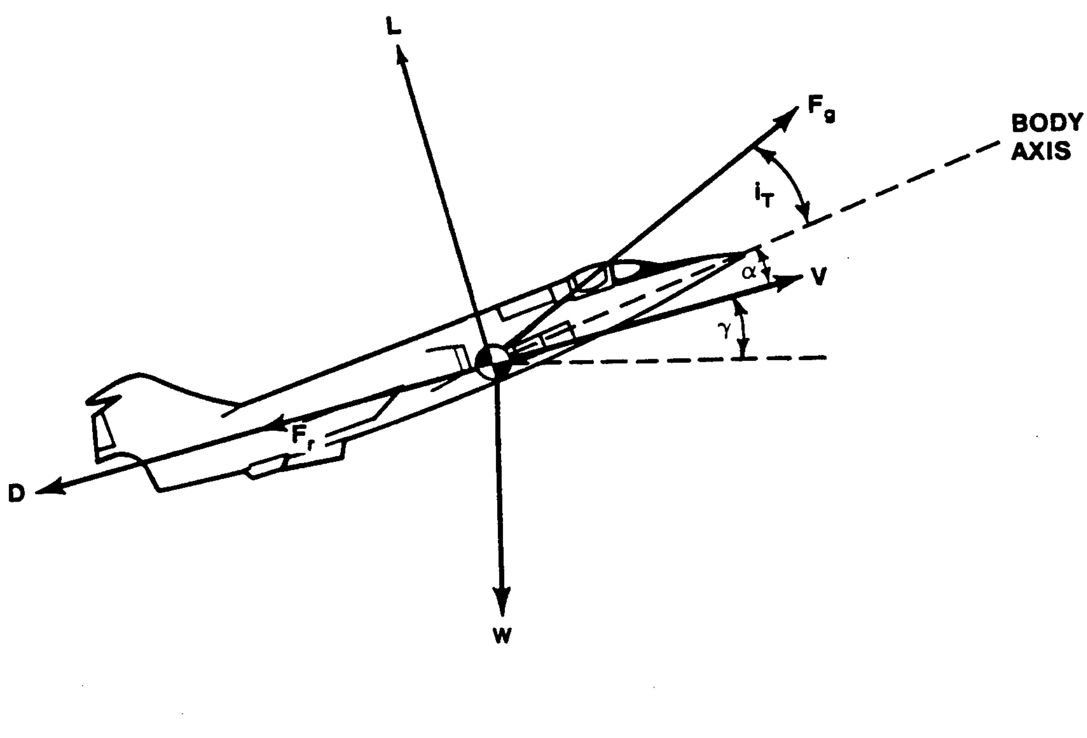

FIGURE 12.3. FORCE DIAGRAM

F_g = Gross thrust

F_r = Ram drag

i_T = Thrust incidence angle

α = Angle of attack

γ = Flight path angle

D = Drag

L = Lift

W = Weight

V = True airspeed

Summing Forces along the flight path (velocity vector)

$$\Sigma F_x = F_g \cos (i_T + \alpha) - F_r - D - W \sin \gamma = \frac{W}{g} \dot{V}$$

Defining

$$F_n = F_g \cos (i_T + \alpha) - F_r$$

$$a_x = \dot{V} + g \sin \gamma$$

Then

$$F_n - D = F_{ex} = \frac{W}{g} a_x \tag{12.46}$$

Similarly, perpendicular to the flight path,

$$\Sigma F_z = L + F_g \sin (i_T + \alpha) - W \cos \gamma = \frac{W}{g} V \dot{\gamma}$$

Defining normal acceleration,

$$a_z = V \dot{\gamma} + g \cos \gamma$$

Then

$$L + F_g \sin (i_T + \alpha) = \frac{W}{g} a_z \tag{12.47}$$

Note that Equations 12.46 and 12.47 make no assumptions concerning steady or non-steady flight. These equations are the basis for the in-flight measurement of F_{ex} by the use of flight path accelerometers or inertial systems.

Specific excess power is

$$P_s = \frac{F_{ex}V}{W} = \dot{H} + V\frac{\dot{V}}{g} \qquad (12.48)$$

where $\dot{H}$ is the true (tapeline) altitude rate of change, and V is the inertial velocity. This equation is fundamental to data reduction for many of the flight test methods, e.g.

Level, unaccelerated flight: $F_{ex} = 0$

Constant airspeed climbs/descents: $\dot{H} = F_{ex}\, V/W$

Climb potential from level accels: $\dot{H} = \frac{F_{ex}V}{W} - V\frac{\dot{V}}{g}$

Any performance parameter can be determined by knowing F_{ex} for an arbitrary set of flight conditions. This is the parameter that is most easily corrected to determine standard day performance. Therefore, F_{ex} is the basic performance parameter which must be determined from flight test.

12.5.2 Determination of F_{ex}

Classically, Equation 12.48 determined excess thrust from flight test data. By recording pitot-static data (H_i and V_i) during a controlled maneuver, test day F_{ex} may be calculated directly. This method is inexpensive, reliable, but subject to many errors: instrument lag and hysteresis, the results of imprecise flying, and the need for precise temperature measurement. Additionally, time measurement must be very precise since rates must be calculated from difference equations, i.e.

$$\dot{V} = \frac{\Delta V}{\Delta t}, \quad \dot{H} = \frac{\Delta H}{\Delta t}$$

Another approach is the use of flight path accelerometers or inertial systems to measure a_x and a_z directly. These systems are accurate and do not require precise flying, but they are expensive and require precise measurement of angle of attack (to determine direction of the flight path). Upwash, vibrations, and nose boom bending can cause large errors.

Finally, H and V may be determined by radar or cinetheodolite tracking. The accuracy of this method depends on equipment accuracy, range, and atmospheric conditions (winds, pressure levels, etc.)

The first (pitot-statics) method is the most widely used and is used at the TPS.

12.5.3 Correction to Standard Conditions

There are several approaches to data reduction. They are similar in that they correct the flight test data for nonstandard conditions, essentially predicting the performance of the aircraft flown on a standard day at the standard altitude and at the standard weight. They differ primarily in the sequence and method in which the corrections are applied. One method is to apply corrections in a "one-at-a-time" sequence. This step-by-step method clearly delineates the assumptions and approximations used to reduce the data, but must be modified for each type of test to be analyzed. An example of this method is presented below for both climb and descent performance.

A more general approach is the method of standardizing excess thrust. This method, used in the Test Pilot School's data reduction system, is described following the step-by-step method.

12.5.4 Climb Performance Data Reduction Using Step-By-Step Method

12.5.4.1 General. Once a climb speed schedule has been obtained, using any of the methods discussed in Chapter 9, the flight test program will call for a series of climbs to be performed to determine the following:

Rate of climb (or time to climb)

Fuel used during the climb

Distance traveled in climb

In order to permit meaningful comparison between flight test data performed under varying conditions, it is necessary to correct all values to some standard condition. Normally, all climb performance data are reduced to ICAO standard day conditions for comparison and presentation.

The fuel used and the distance traveled during the climb are, to a large degree, dependent upon the rate of climb. Therefore, the major emphasis is devoted to adjustments of the rate of climb.

Seven corrections are required in our analysis. Four are associated with temperature variables, one corrects for wind, and two result from nonstandard weight differences. The corrections are listed below in the order in which they are generally applied to climb data:

Correction	Cause
1. Tapeline altitude	Nonstandard temperature
2. True speed	Nonstandard temperature
3. Thrust	Nonstandard temperature
4. Wind	Wind gradient
5. Acceleration	Nonstandard temperature lapse rate
6. Inertia	Nonstandard weight
7. Induced drag	Nonstandard weight

In the following analysis the numerical rate of climb subscripts will maintain their identity from beginning to end. Subscript "t" will refer to test day uncorrected climb conditions. Subscript "1" will refer to values corrected for tapeline altitudes only. Each succeeding subscript will refer to values adjusted for all preceding corrections. The subscript "s" will refer to standard day data with corrections applied.

12.5.4.2 Tapeline Altitudes. With the aircraft altimeter set to 29.92" Hg, accurate values of H_c, and therefore of P_a, are available from climb data. But a given change in P_a on a test day does not represent the change in true, or "tapeline," altitude that it would represent on a standard day, since the change of pressure with altitude is not a linear function. Figure 12.4 illustrates the pressure versus altitude relationship for a standard day and for a hotter than standard day.

Let us assume that during a climb on a hotter than standard day our altimeter sensed a change of pressure of ΔP. Since the altimeter is constructed on a standard pressure lapse rate, this ΔP will register a change of altitude of ΔH_i on the instrument. However, we are actually on the nonstandard pressure curve and our actual change of altitude would be ΔH_a. As shown in Figure 12.4, this actual altitude, or true altitude change, is greater than registered by the altimeter. The additional energy for this increased altitude must be provided by the engine, and a correction is required to determine the standard day rate of climb.

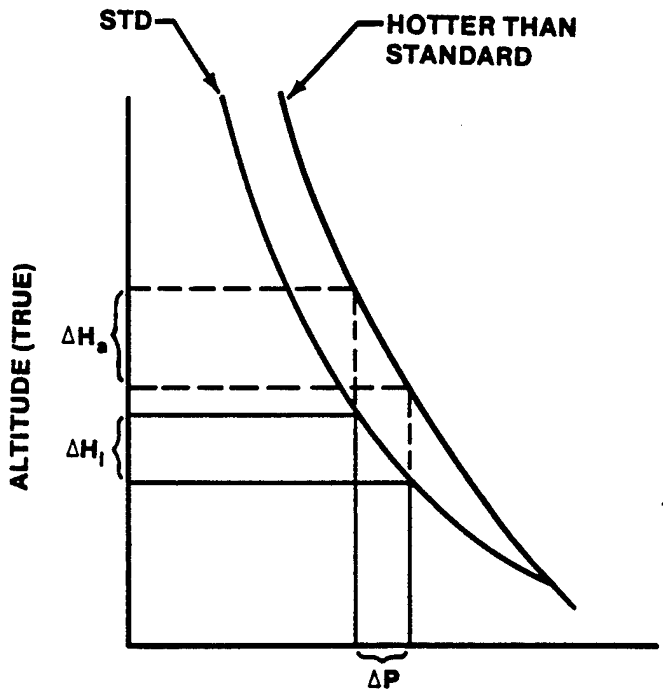

FIGURE 12.4. PRESSURE (ALTIMETER READING)

Since we fly <u>pressure</u> <u>altitudes</u>,

$$P_t = P_s \text{ (as sensed by altimeter)}$$

$$P_t = \rho_t \, gRT_t \qquad P_s = \rho_s \, gRT_s$$

Therefore

$$\rho_t \, gRT_t = \rho_s \, gRT_s$$

or

$$\frac{\rho_s}{\rho_t} = \frac{T_t}{T_s}$$

Recall from Chapter 5 that

$$\frac{dH}{dP} = -\frac{1}{\rho g}$$

Then, for small changes,

$$\Delta H = -\frac{\Delta P}{\rho g}$$

For a given ΔP (as sensed by the altimeter)

$$\text{Indicated or apparent } \Delta H_i = -\frac{\Delta P}{\rho_s g}$$

$$\text{Actual } \Delta H_a = -\frac{\Delta P}{\rho_t g}$$

Now dividing ΔH_a by ΔH_i

$$\frac{\Delta H_a}{\Delta H_i} = \frac{\rho_s}{\rho_t} \quad \text{or} \quad \Delta H_a = \Delta H_i \frac{\rho_s}{\rho_t}$$

Since

$$\frac{\rho_s}{\rho_t} = \frac{T_t}{T_s}$$

and dividing by Δt

$$\frac{\Delta H_a}{\Delta t} = \frac{\Delta H_i}{\Delta t} \frac{T_t}{T_s}$$

where $\Delta H_i/\Delta t$ is the apparent rate of climb (R/C_t) and $\Delta H_a/\Delta t$ is the actual rate of climb (R/C)

This can be written as

$$R/C_1 = R/C_t \frac{T_t}{T_s} \tag{12.49}$$

This is called the tapeline altitude correction. This correction is always applied to climb data. Temperatures are absolute and are taken at the midpoint of the altitude band under consideration.

12.5.4.3 True Speed and Thrust Correction. The nonstandard temperature effects on true speed and thrust are so closely related that they cannot be easily separated. A change in temperature will produce a change in thrust and in true speed, V, and the change in V will, in turn, produce a secondary change in thrust of a jet engine or in thrust horsepower of a piston engine or turboprop. Ultimately, the thrust correction must be based upon known thrust data, which is obtained from manufacturer's thrust stand data for the particular engine. For simplicity, the two effects (true speed and thrust) will be analyzed simultaneously and separated into the two causes after analysis. The analysis is based upon two premises:

1) Thrust horsepower available (THP_a) changes with true velocity and with thrust.

2) Thrust horsepower required (THP_r) or drag, changes with true velocity, but not with thrust. The only effect of temperature on drag is assumed to be through the change in V.

Assuming an unaccelerated climb and recalling the energy relationships

$$R/C = \frac{(F_n - D)\ V}{W}$$

$$R/C = \frac{F_n V - DV}{W}$$

$$F_n V = THP_a \text{ and } DV = THP_r$$

Then

$$\frac{THP_{a_s}}{THP_{a_t}} = \frac{F_{n_s} V_s}{F_{n_t} V_t}$$

Since $M_t = M_s$ and $V = M\sqrt{\gamma gRT} = KM\sqrt{T}$

$$THP_{a_s} = THP_{a_t}\left(\frac{F_{n_s}}{F_{n_t}}\sqrt{\frac{T_s}{T_t}}\right) \tag{12.50}$$

Similarly

$$THP_{r_s} = THP_{r_t}\left(\frac{D_s}{D_t}\sqrt{\frac{T_s}{T_t}}\right) \tag{12.51}$$

From aerodynamic theory, drag is given by

$$D = 1481\ \delta M^2 SC_D$$

Assuming test and standard day C_D are equal (while this is not true, this correction will be made in step 7), and since $M_s = M_t$, then

$$\frac{D_s}{D_t} = \frac{\delta_s}{\delta_t} \tag{12.52}$$

Then

$$R/C_3 = \frac{THP_{a_s} - THP_{r_s}}{W}$$

$$= \left[\frac{F_{n_s}}{F_{n_t}}\sqrt{\frac{T_s}{T_t}}\,THP_{a_t} - \frac{\delta_s}{\delta_t}\sqrt{\frac{T_s}{T_t}}\,THP_{r_t}\right]\frac{1}{W} \tag{12.53}$$

$$= \frac{\delta_s}{\delta_t}\sqrt{\frac{T_s}{T_t}}\left[\frac{\left(\frac{F_{n_s}}{\delta_s}\right)}{\left(\frac{F_{n_t}}{\delta_t}\right)}THP_{a_t} - THP_{r_t}\right]\frac{1}{W} \tag{12.54}$$

Let

$$\frac{F_{n_s}}{\delta_s} = \frac{F_{n_t}}{\delta_t} + \Delta \frac{F_n}{\delta}$$

Then

$$\frac{\left(\frac{F_{n_s}}{\delta_s}\right)}{\left(\frac{F_{n_t}}{\delta_t}\right)} = \frac{\left(\frac{F_{n_t}}{\delta_t}\right) + \left(\Delta \frac{F_n}{\delta}\right)}{\left(\frac{F_{n_t}}{\delta_t}\right)} = 1 + \frac{\left(\Delta \frac{F_n}{\delta}\right)}{\left(\frac{F_{n_t}}{\delta_t}\right)} \tag{12.55}$$

$$R/C_3 = \frac{\delta_s}{\delta_t}\sqrt{\frac{T_s}{T_t}} \left[\left(1 + \frac{\left(\Delta \frac{F_n}{\delta}\right)}{\left(\frac{F_{n_t}}{\delta_t}\right)} \right) THP_{a_t} - THP_{r_t} \right] \frac{1}{W}$$

$$= \frac{\delta_s}{\delta_t}\sqrt{\frac{T_s}{T_t}} \left[THP_{a_t} - THP_{r_t} \right] \frac{1}{W} + \frac{\delta_s}{\delta_t}\sqrt{\frac{T_s}{T_t}} \frac{THP_{a_t}}{W} \frac{\left(\Delta \frac{F_n}{\delta}\right)}{\left(\frac{F_{n_t}}{\delta_t}\right)}$$

$$= \frac{\delta_s}{\delta_t}\sqrt{\frac{T_s}{T_t}} R/C_1 + \frac{\delta_s}{\delta_t}\sqrt{\frac{T_s}{T_t}} \frac{THP_{a_t}}{W} \frac{\left(\Delta \frac{F_n}{\delta}\right)}{\left(\frac{F_{n_t}}{\delta_t}\right)} \tag{12.56}$$

For convenience, the second term of Equation 12.56 is designated $\Delta R/C_1$ and the equation becomes

$$R/C_3 = \frac{\delta_s}{\delta_t}\sqrt{\frac{T_s}{T_t}} R/C_1 + \Delta R/C_1 \tag{12.57}$$

Equation 12.57 is now the total correction for tapeline altitude, true speed, and thrust.

12.5.4.4 $\Delta R/C_1$ Determination. Values of $\Delta\ F_n/\delta$ must come from thrust data, and are often presented in chart form. For jet aircraft, a common approach is to organize the charts so that a plot will be entered with Mach, engine speed, N_t, and test day temperature, T_t, to get the generalized thrust parameter, $(F_n/\delta)_t$.

The same chart is entered at N_s and T_s to get a standard day thrust parameter, $(F_n/\delta)_s$

Then

$$\left(\Delta \frac{F_n}{\delta}\right) = \left(\frac{F_n}{\delta}\right)_s - \left(\frac{F_n}{\delta}\right)_t$$

Note that five variables, temperature, RPM, altitude, speed, and weight, go into the determination of the thrust effect $\Delta R/C_1$. If thrust stand data is available in the correct form at each speed, altitude, and temperature flown, then a simpler form of the $\Delta R/C_1$ equation can be used.

Substituting

$$\sqrt{\frac{T_s}{T_t}} = \frac{V_s}{V_t} \text{ and } THP_a = F_n V$$

$$\Delta R/C_1 = \frac{\delta_s}{\delta_t}\frac{V_s}{V_t}\frac{F_{n_t}}{W}\frac{V_t\left(\Delta \frac{F_n}{\delta}\right)}{\left(\frac{F_{n_t}}{\delta_t}\right)}$$

$$\Delta R/C_1 = \frac{V_s \delta_s}{W}\left(\Delta \frac{F_n}{\delta}\right) \qquad (12.58)$$

$\Delta R/C_1$ is usually a large correction and is always applied to nonstandard climb data.

12.5.4.5 Wind Correction. A wind of constant velocity will not affect the rate of climb of an aircraft regardless of its magnitude or direction. However, it is normal to experience some wind gradient in a climb even under

the best of conditions. A wind gradient affects the climb performance in two ways. First, if an aircraft is climbing into an increasing headwind (called a positive gradient), its inertia will carry it along at essentially a constant inertial speed. But the increased headwind velocity will register on the pitot-static instruments as an increase in airspeed. The pilot will correct the airspeed by raising the nose of the aircraft, and the rate of climb will increase.

Secondly, if the wind direction changes, the relative wind vector is rotated and the effect on the pitot-static instruments is similar to a wind gradient. Since the mechanics of calculating this effect are quite detailed, and because the effect is usually much less than the gradient effect, it will not be considered during this discussion.

The term V_W will be used to denote the total resultant headwind/tailwind component. The wind gradient with altitude is equal to dV_W/dH. For purpose of analysis, dV_W/dH may be treated as a sudden acceleration equal in magnitude to the change in wind velocity. From the energy equations

$$\frac{(F_n - D)\ V}{W} = \frac{dH}{dt} + \frac{V}{g}\frac{dV}{dt}$$

Assuming an unaccelerated standard day climb

$$R/C_s = R/C_t + \frac{V}{g}\frac{dV}{dt}$$

Expanding dV/dt into dV/dH dH/dt

$$R/C_s = R/C_t + \frac{V}{g}\frac{dV}{dH}\frac{dH}{dt}$$

$$= R/C_t + \frac{V}{g}\frac{dV}{dH}\ R/C_t \qquad (12.60)$$

The quantity dV/dH is produced by, and is equal in magnitude to, the vertical wind gradient, dV_W/dH, but has the opposite sign. This is due to the difference in sign conventions between wind velocities and aircraft velocities. If we maintain the convention of a headwind being positive, the equation becomes

$$R/C_s = R/C_t - \frac{V}{g}\frac{dV_W}{dH} R/C_t \tag{12.61}$$

Recalling that we make corrections in the proper order and that we must always use test R/C corrected for all previous nonstandard conditions, the equation becomes

$$R/C_4 = R/C_3 - \frac{V}{g}\frac{dV_W}{dH} R/C_3 \tag{12.62}$$

where R/C_3 is the test rate of climb corrected for tapeline altitude, true speed, and thrust, and R/C_4 includes the wind correction.

As for most corrections, the wind correction is less valid if the error is large. Attempts are made to minimize the error by flying 90° to the wind direction, by flying in light winds, and by flying successive climbs in opposite directions when possible. This last procedure also serves as a check on the magnitude of wind error, since the vertical displacement of the R/C curve will be a direct measure of the wind gradient (Figure 12.5).

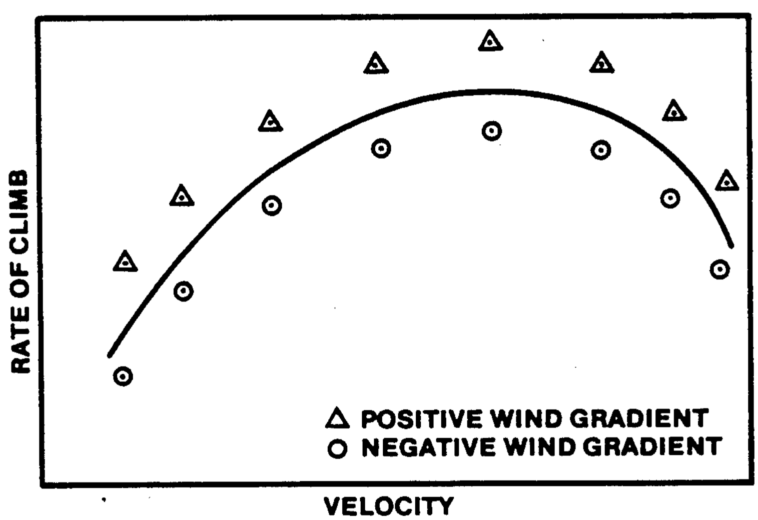

FIGURE 12.5. WIND GRADIENT EFFECT ON RATE OF CLIMB

Since winds are generally not constant in speed or direction during climbs, there will normally remain some residual error. The valid application of the wind correction equation requires that this wind gradient be known accurately. This requirement seriously limits the effective use of this correction. The release of a weather balloon at the time the climb is being performed will give fair, but far from perfect wind data. Normally, the correction equation is not used. Instead, the most widely used technique is to perform a number of climbs in different directions (90° to the wind, if possible) on different days, plot all data points on a single chart and draw an average line through them, and ignore the residual wind error altogether. It is hoped that the error will thus be averaged to a negligible value.

12.5.4.6 Acceleration Error. The first three corrections that were made adjusted the rate of climb for a nonstandard temperature. Compensation was made for a nonstandard true velocity due to nonstandard temperature. Under most actual atmospheric conditions, there will exist not only a temperature difference from standard, but a change in this temperature difference with altitude or a nonstandard temperature lapse rate. This comparison is illustrated in Figure 12.6.

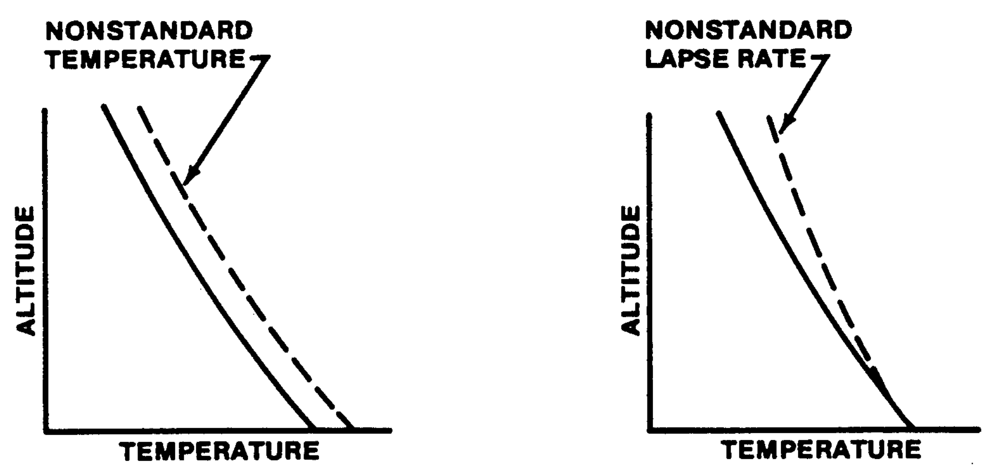

FIGURE 12.6.

Since true velocity changes with temperature (for a constant indicated velocity), this nonstandard temperature gradient will introduce a nonstandard acceleration for which a correction to the rate of climb must be applied.

Considering a climb during which data are taken at altitudes H_1 and H_2, let $\Delta H = H_2 - H_1$, and $\Delta V = V_2 - V_1$, where V_1 is taken at H_1 and V_2 at H_2. To correct this climb to zero acceleration it is only necessary to apply Equation 12.60

$$R/C_s = R/C_t + \frac{V}{g}\frac{dV}{dH} R/C_t$$

where dV/dH is obtained from test day data and is equal to $\Delta V/\Delta H$. However, this will produce the corrected rate of climb only if the desired standard day climb schedule is a constant V (zero acceleration) schedule. Most climb schedules are not constant V schedules, and a slightly different correction equation is required. If the test day acceleration between altitudes H_1 and H_2 is not equal to the desired standard day acceleration, the equation must be modified to

$$R/C_s = R/C_t + \left(\frac{V_t}{g}\frac{\Delta V_t}{\Delta H} - \frac{V_s}{g}\frac{\Delta V_s}{\Delta H}\right) R/C_t \qquad (12.63)$$

assuming that $V_t \cong V_s$, then

$$R/C_s \cong R/C_t - \frac{V_t}{g\Delta H}(\Delta V_s - \Delta V_t)\ R/C_t$$

adjusting this to the usual form

$$R/C_5 \cong R/C_4 - \frac{V_t}{g\Delta H}(\Delta V_s - \Delta V_t)\ R/C_4 \qquad (12.64)$$

where R/C_4 is corrected for tapeline altitude, true speed, thrust, and wind.

The acceleration correction is always applied to climb data and should be applied in incremental altitude bands.

12.5.4.7 Weight Corrections. "Standard weight" is determined by the test force and may be any weight so designated. The standard weight parameter, at any given altitude, should normally be the weight at level off following a

standard day takeoff and climb to that altitude. The standard weight often involves an average of many test weights and may be adjusted, if necessary, as the test program progresses.

Nonstandard weight of an aircraft affects its climb performance in two ways. First, an aircraft which is heavier than normal requires more energy from the engine to increase its altitude a given ΔH, since potential energy increase equals $W\Delta H$. Second, a heavier aircraft must have a higher wing loading to maintain equilibrium; therefore, at a given speed it must fly at a higher angle of attack and will generate more induced drag. This extra drag must also be overcome by the engine.

For purposes of this analysis it will be assumed that the total excess thrust $(F_n - D)$ is expended in generating rate of climb. While this is not strictly true in all cases, the portion of energy spent on acceleration will have little effect on the analysis.

From energy concepts

$$R/C = \frac{(F_n - D)}{W} V = \frac{F_n V - DV}{W}$$

Differentiating this expression with respect to weight

$$\frac{d\,R/C}{dW} = -\frac{(F_n - D)\;V}{W^2} - \frac{V}{W}\frac{dD}{dW}$$

The assumption was made that the small change in angle of attack due to weight will not affect the thrust (i.e. $dF_n/dW = 0$). Then, evaluating these terms at the test condition

$$\frac{d(R/C)}{dW} = -\left(\frac{R/C_t}{W_t} + \frac{V_t}{W_t}\frac{dD}{dW}\right)$$

Multiplying by dW, and using the weight changes $\Delta W = W_s - W_t$ for dW, $\Delta D = D_s - D_t$ for dD, and $\Delta R/C = R/C_s - R/C_t$ for $d(R/C)$

$$\Delta R/C_{weight} = -\left(R/C_t\,\frac{\Delta W}{W_t} + \frac{V_t}{W_t}\,\Delta D\right) \qquad (12.65)$$

The first term in the parentheses in the above equation is the effect of increased potential energy required for heavier aircraft, and is called the "inertia effect". The inertia correction by convention is called $\Delta R/C_2$ and is given by

$$\Delta R/C_2 = R/C_t \frac{(W_t - W_s)}{W_t} \tag{12.66}$$

$$\Delta R/C_2 = R/C_5 \frac{(W_t - W_s)}{W_t} \tag{12.67}$$

where R/C_5 incorporates all previous corrections.

The "induced drag" portion of the weight correction is derived from an equation for induced drag in terms of known aircraft parameters as follows:

$$D_i = \frac{K(nW)^2 \cos^2 \gamma}{b^2 e M^2 \delta} \tag{12.68}$$

when K is a dimensional constant.

The change in induced drag caused by weight is then

$$\Delta D_i = \frac{K \cos^2 \gamma}{b^2 M^2 e\ \delta_t} \left(W_s^{\ 2} - W_t^{\ 2} \right) \tag{12.69}$$

12.5.4.8 Summary. Using the above information, the total standard day correction for climb performance may be summarized as

$$R/C_1 = (R/C_t) \frac{T_t}{T_s}$$

$$\Delta R/C_1 = \frac{V_s\ \delta_s}{W_t} \left(\frac{F_{n_s}}{\delta_s} - \frac{F_{n_t}}{\delta_t} \right)$$

$$R/C_3 = R/C_1 \frac{\delta_s}{\delta_t} \sqrt{\frac{T_s}{T_t}} + \Delta R/C_1$$

$$R/C_4 = R/C_3 - \frac{V_t}{g}\frac{dV_W}{dH} R/C_3$$

$$R/C_5 = R/C_4 - \frac{V_t}{g}\left(\frac{\Delta V_s}{\Delta H} - \frac{\Delta V_t}{\Delta H}\right) R/C_4$$

$$\Delta R/C_{weight} = R/C_5 \frac{(W_t - W_s)}{W_t} + \frac{V_t}{W_t}\frac{K \cos^2\gamma}{b^2 M^2 e\delta_t}(W_t^2 - W_s^2)$$

$$R/C_S = R/C_5 + \Delta R/C_{weight}$$

Applying the corrections in this order should result in the larger corrections being applied first to minimize errors. This may not always occur, particularly if the test weight varies considerably from standard. But in most cases, climb tests are made directly from takeoff and weight corrections can be kept small.

12.5.5 Descent Performance Data Reduction Using Step-By-Step Method

Descent performance can be analyzed in exactly the same manner as climb performance, and the same equations apply. Obviously, a negative rate of climb will result, which can be considered as a rate of descent if desired. Corrections to descent performance are not always as valid as those to climb performance, however.

12.5.5.1 Thrust Correction. Thrust data is usually complete for engines operating at military or maximum power. At idle power, however, data is not complete, and the engine trim is often less reliable. Since the thrust is usually relatively small at idle, it is often a good procedure to simply consider it zero and apply no thrust correction. Of course, an increase in thrust will decrease the rate of descent.

12.5.5.2 Weight Correction. In a descent, the induced drag portion of the weight correction will remain as in the climb. The inertia portion is different in a descent since the forward component of force acting on the aircraft is primarily a component of weight.

A change in weight will result in a change in induced drag but also in the component opposing drag. If the speed is held constant the result may be an increase or a decrease in rate of descent depending upon whether the glide speed is above or below that for best L/D ratio. If the best L/D is maintained, the induced drag will remain constant, and the glide angle will also remain very nearly constant, but the rate of descent will increase.

In flight test operations descents can usually be made at or near the desired standard descent weight, and the weight ambiguity can usually be neglected.

12.5.6 Standardization of Excess Thrust.

While the step-by-step method clearly delineates each assumption and approximation to correct climb performance data to standard day conditions, it is extremely laborious and must be modified for each type of test. A simpler, and more general technique is to standardize the excess thrust to standard day conditions. This allows each energy test (level accel, sawtooth climb, check climb, descent, and turns) to be analyzed using a similar approach.

The first step is to calculate the test day excess thrust using the expression

$$F_{ex_t} = \frac{W_t}{V_t}\left(\dot{H}_t + \frac{V_t \dot{V}_t}{g}\right) \qquad (12.70)$$

where V_t is the average true airspeed on the test day, $\dot{V}_t$ is its rate of change at the instant under investigation, and $\dot{H}_t$ is the rate of change of the aircraft's tapeline altitude. This can be obtained from calibrated altitude, using the same tapeline altitude correction derived in the step by step approach, that is

$$\dot{H}_t = \dot{H}_c\left(\frac{T_t}{T_s}\right) \qquad (12.71)$$

The second step is to estimate, or predict, what the excess thrust would have been on a standard day by using the relationship

$$F_{ex_s} = F_{ex_t} + \Delta F_n - \Delta D \qquad (12.72)$$

The terms ΔF_n and ΔD represent the predicted change in the net thrust and drag, respectively, between the test day and standard day.

$$\Delta F_n = F_{n_s} - F_{n_t} \tag{12.73}$$

$$\Delta D = D_s - D_t$$

As with the step-by-step method, the change in the net thrust is calculated by using engine thrust data provided by the manufacturer. Test day and standard day drag are calculated by using the relationship

$$D = 1481\,\delta\,M^2 S C_D$$

where δ and M are measured in the flight test and C_D can be calculated from C_L, using the aircraft's drag polar. The aircraft lift coefficient, C_L, is calculated from the relationship

$$C_L = \frac{nW}{1481\delta M^2 S}$$

where n and W are measured quantities.

At first glance the need for both the engine thrust data and the aircraft's drag polar to perform the data reduction appears to be a ridiculous requirement. Given thrust and drag, the performance characteristics being tested could be calculated. So why bother with the test? The answer lies in how this data is used. While test day net thrust and drag are calculated from the thrust data and drag polar, respectively, they are not used in an absolute sense. They are used only to obtain correction terms. That is, test day net thrust is not used by itself but only in conjunction with standard day net thrust to calculate the delta change. As a result, small errors in the thrust data have little effect on the accuracy of the data reduction, provided standard day and test conditions are nearly identical. This is because bias error will be present in both and should cancel. This is in contrast to the accuracy requirement for predicting aircraft performance from thrust data and drag polar, without any test data. In this case, errors in either will translate directly into errors in predicted excess thrust and therefore

aircraft performance. In summary, while data reduction requires thrust curves and a drag polar, these data can be slightly in error (i.e. estimated data) without significantly affecting the quality of the flight test data reduction.

The third, and final step, is to use the predicted excess thrust to calculate the other quantities of interest. For example, standard day specific excess power can be calculated from

$$P_{s_s} = \frac{F_{ex_s} V_s}{W_s} \tag{12.75}$$

and standard day climb performance can be calculated from

$$\dot{H}_s = \frac{F_{ex_s} V_s}{W_s} - \frac{V_s \dot{V}_s}{g} \tag{12.76}$$

where $\dot{V}_s$ represents the rate of change of true airspeed associated with the desired climb schedule.

12.5.7 Level Acceleration and Sawtooth Climb Data Reduction

The level acceleration and sawtooth climb tests are used to gather data to determine P_s and to predict sustained turn capability. From a data reduction point of view the two techniques are similar, only differing in the magnitude of the altitude rate versus airspeed rate terms. A time history of the following parameters is the required input to the data reduction.

Altitude, H_i (ft)

Indicated Airspeed, V_i (ft/sec)

Engine RPM, N (RPM)

Outside Air Temperature, T_a (^oK)

Aircraft Weight, W_t (lbs)

In addition, the altitude at which the data are to be standardized, as well as the corresponding aircraft's standard weight, W_s (lb) are required. The following calculations should then be performed.

a. Use pitot-static relationships to calculate

$$H_{c_1},\ V_{T_1},\ T_{a_1},\ M_1$$

$$H_{c_2},\ V_{T_2},\ T_{a_2},\ M_2$$

where the subscripts 1 and 2 refer to two adjacent data points, e.g. two different speeds for level acceleration tests or two altitudes for a sawtooth climb.

b. Calculate test day average values

$$H_c = \frac{H_{c_1} + H_{c_2}}{2}$$

$$M = \frac{M_1 + M_2}{2}$$

$$T_t = \frac{T_{a_1} + T_{a_2}}{2}$$

$$W_t = \frac{W_{t_1} + W_{t_2}}{2}$$

$$N = \frac{N_1 + N_2}{2}$$

c. Use pitot-static relationships to calculate the following test and standard day parameters.

V_{T_t}, δ_t, θ_t calculated from H_c, M, T_t

V_{T_s}, δ_s, θ_s from M and standard altitude

d. Calculate rates

$$\dot{H}_c \cong \frac{H_{c_2} - H_{c_1}}{\Delta t} \tag{12.78}$$

$$\dot{V}_{T_t} \cong \frac{V_{T_2} - V_{T_1}}{\Delta t} \tag{12.79}$$

where Δt is the recorded time difference between data points 1 and 2.

e. Calculate test day parameters

$$F_{ex_t} = W_t \left[\frac{\dot{V}_{T_t}}{g} + \frac{\dot{H}_c}{V_{T_t}} \; \frac{T_t}{T_s} \right]$$

where T_s is the standard day ambient temperature at the test altitude

$$\gamma_t = \sin^{-1} \left[\frac{\dot{H}_c}{V_{T_t}} \; \frac{T_t}{T_s} \right]$$

(12.80)

$$C_{L_t} = \frac{W_t \cos \gamma_t}{1481 \; M^2 S \; \delta_t}$$

C_{D_t}: From drag curve using C_{L_t} and M

$$D_t = (1481 \; M^2 S \; \delta_t) \; C_{D_t}$$

F_{n_t}: From thrust curve using M, N, θ_t, and δ_t

f. Calculate standard day parameters

$$C_{L_s} = \frac{W_s}{1481 \; M^2 S \; \delta_s}$$

C_{D_s}: From drag curve using C_{L_s} and M

$$D_s = 1481\ M^2 S\ \delta_s\ C_{D_s} \tag{12.81}$$

F_{n_s}: From thrust curve using M, N, θ_s, and δ_s

$$\Delta D = D_s - D_t$$

$$\Delta F_n = F_{n_s} - F_{n_t}$$

$$F_{ex_s} = F_{ex_t} + \Delta F_n - \Delta D$$

g. Calculate standard day specific excess power, P_s

$$P_{s_s} = F_{ex_s} \frac{V_{T_s}}{W_s} \tag{12.82}$$

h. Predict the aircraft's sustained turn capability by assuming that lift can be increased until the increased drag balances the calculated standard day excess thrust.

$$C_{D_{lim_s}} = C_{D_s} + \frac{F_{ex_s}}{1481\ M^2 S\ \delta_s}$$

$C_{L_{lim_s}}$: From drag curve using $C_{D_{lim_s}}$ and M

$$n_{lim_s} = \frac{C_{L_{lim_s}}}{C_{L_s}}$$

The standard day specific excess power, P_s, should then be plotted versus Mach for a specific standard altitude and power setting. A family of these curves, for various altitudes, is shown in Figure 12.7:

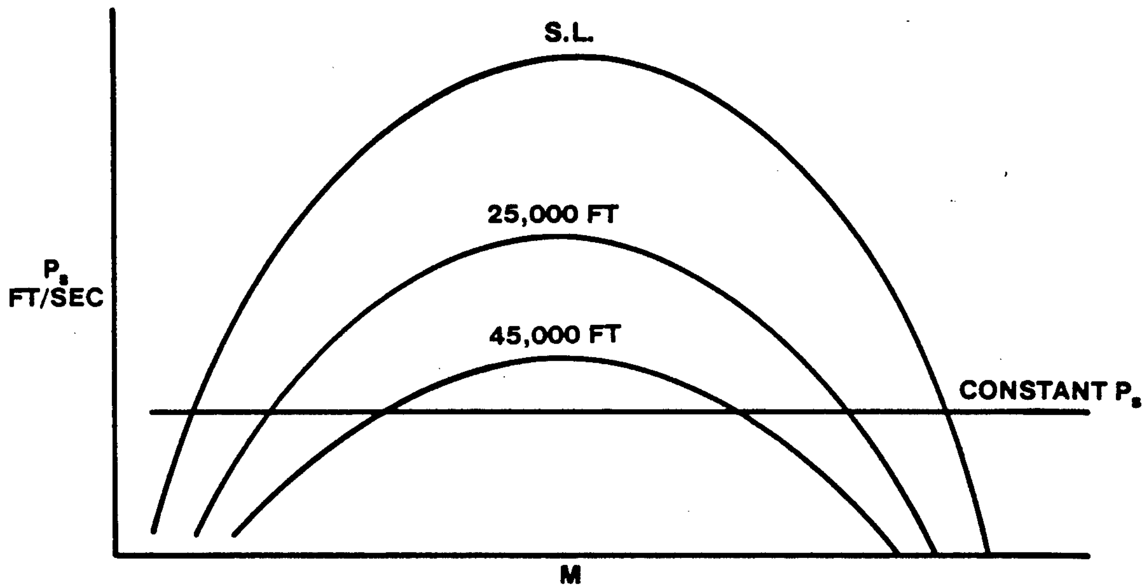

FIGURE 12.7. P_s VERSUS M FROM LEVEL ACCEL

The contour plot of constant P_s as a function of altitude and Mach (Figure 12.8) can now be generated. By drawing lines at constant P_s and reading off the Mach corresponding to each altitude, the following crossplot is produced.

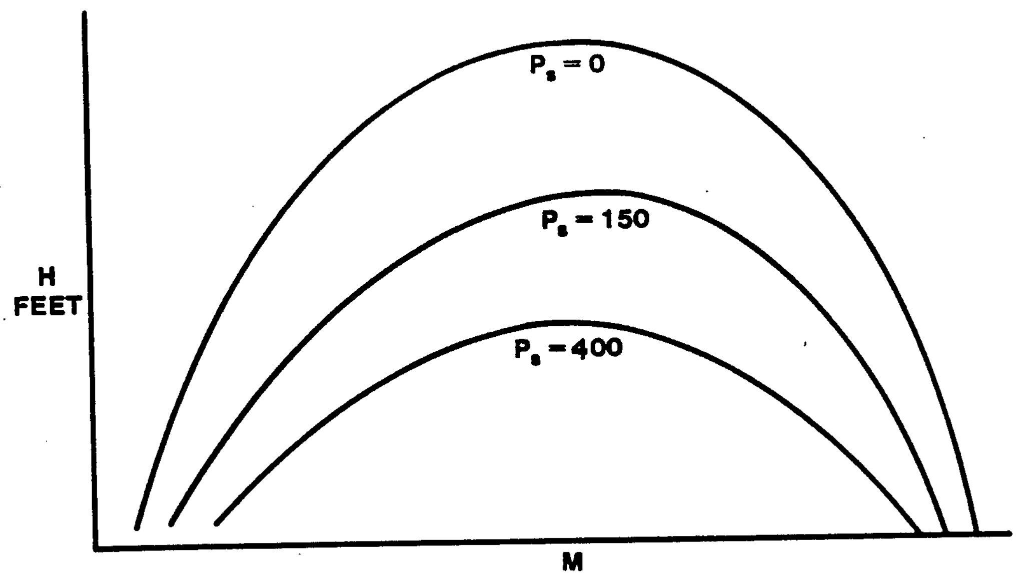

FIGURE 12.8. P_S VERSUS H AND M FROM LEVEL ACCEL

On this chart, the climb schedule for maximum rate of climb can be found by a line joining the peaks of the curves. Lines of constant specific energy may also be drawn and the points where these are tangent to the lines of constant P_S will define the optimum energy climb schedule (Figure 12.9).

In practice, it is easier and almost as accurate to obtain an approximation of the optimum energy climb schedule. This is done by selecting an altitude one percent below the peak of any P_S curve and finding a point at this altitude on the high speed side. Joining these points will give a climb schedule which, for the subsonic case, will usually agree with the optimum energy climb within the accuracy of data obtained. A supersonic climb schedule may also be found from this plot, although approximate methods may not be effective.

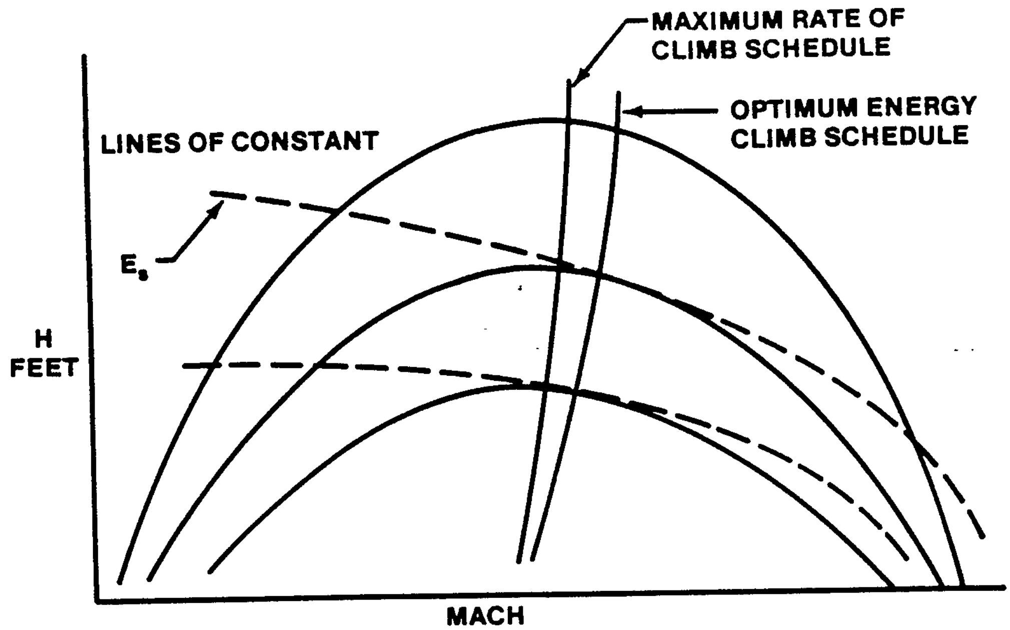

FIGURE 12.9. CLIMB SCHEDULES FROM LEVEL ACCEL DATA

The maximum sustainable load factor, n_{lim_s}, predicted from level acceleration data should be plotted versus Mach for a specific altitude and and power setting. This data should be compared with the data generated during the turn performance test to assure agreement. But since the standard weight for level acceleration tests is frequently different than that for turn performance, care should be taken to use a common standard weight when comparing these two methods.

12.5.8 Check Climb Data Reduction

A series of check climbs are flown to verify and refine the optimum climb schedule predicted by the level acceleration tests. It is important that the pilot accurately fly the prescribed schedule since the data reduction technique described below uses the test day climb schedule in the standard day calculations. To analyze check climb data, a time history of the following parameters should be recorded:

Altitude, H_i (ft)

Indicated Airspeed, V_i (ft/sec)

Engine RPM, N (RPM)

Outside Air Temperature, T_a (^oK)

Aircraft Weight, W_t (lbs)

The standard weight, W_{s_0} (lbs) at the initial altitude of the check climb is required.

Check climb data is analyzed by breaking the climb into a series of small altitude bands, with the previous list of parameters recorded at the bottom and the top of each band. Pairs of data points are analyzed to predict the standard day rate of climb, time to climb, fuel used, distance traveled, and aircraft weight for that data band. The cumulative standard day time, fuel, distance, and weight are then calculated by summing the time, fuel used, and distance for each of the individual data bands.

a. For the "n+1" data band, use pitot-static relationships to calculate:

$$H_{c_1}, V_{T_1}, T_{a_1}, M_1$$

$$H_{c_2}, V_{T_2}, T_{a_2}, M_2$$

where the subscripts 1 and 2 refer to the bottom and top data points, respectively, of the data interval. The data reduction technique calculates the standard day rate of climb, based upon this test data, for an altitude and Mach that is the average of the test day values.

b. Calculate test day average values:

$$H_c = \frac{H_{c_1} + H_{c_2}}{2} \tag{12.77}$$

$$M = \frac{M_1 + M_2}{2}$$

$$T_t = \frac{T_{a_1} + T_{a_2}}{2}$$

$$W_t = \frac{W_{t_1} + W_{t_2}}{2}$$

$$N = \frac{N_1 + N_2}{2}$$

c. Use pitot-static relationships to calculate the following test and standard day parameters:

V_{T_t}, δ_t, θ_t calculated from H_c, M, T_t

V_{T_s}, δ_s, θ_s calculated from M and the standard altitude. For check climb data, the standard altitude is assumed to be the same as the test day pressure altitude, and therefore:

$\delta_s = \delta_t$ (hereafter referred to as simply δ)

d. Calculate rates

$$\dot{H}_c \cong \frac{H_{c_2} - H_{c_1}}{\Delta t_t} \tag{12.78}$$

$$\dot{V}_{T_t} \cong \frac{V_{T_2} - V_{T_1}}{\Delta t_t} \tag{12.79}$$

$$\dot{W}_t \cong \frac{W_{t_2} - W_{t_1}}{\Delta t_t} \tag{12.83}$$

where Δt_t is the recorded time difference between data points 1 and 2.

e. Calculate test day parameters

$$\dot{H} = \dot{H}_c \frac{T_t}{T_s}$$

where T_s is the standard day ambient temperature at the test altitude.

$$F_{ex_t} = W_t \left[\frac{\dot{V}_{T_t}}{g} + \frac{\dot{H}}{V_{T_t}} \right]$$

$$\gamma_t = \sin^{-1}\left(\frac{\dot{H}}{V_{T_t}} \right) \qquad (12.80)$$

$$C_{L_t} = \frac{W_t \cos \gamma_t}{1481\ M^2 S\ \delta}$$

C_{D_t}: From drag curve using C_{L_t} and M

$$D_t = 1481\ M^2 S\ \delta\ C_{D_t}$$

F_{n_t}: From thrust curve using M, N, θ_t, and δ

f. Calculate standard day parameters:

F_{n_s}: From thrust curve using M, N, θ_s, and δ

$$\dot{W}_s = \dot{W}_t + \Delta\dot{W}$$

where $\Delta\dot{W}$ is the difference in the engine fuel flow between the test day and the standard day. This can be calculated by using manufacturer's charts of the engine characteristics. If the engine is a single spool jet engine, the fuel flow should be described by the following functional relationship, as described in Chapter 7.

Corrected fuel flow = function of Mach and corrected RPM

$$\frac{\dot{W}}{\delta\sqrt{\theta}} = f(M, N/\sqrt{\theta})$$

If fuel flow is of this form, then ΔW can be calculated from M, N, δ (since $\delta_s = \delta_t$), θ_s, and θ_t.

Since rate of climb depends upon the aircraft standard weight, and standard weight depends upon time to climb, an iterative approach is necessary to complete the check climb data reduction. Certain variables must be initialized and are then updated each iteration through the loop.

$$\gamma = \gamma_t$$

$$\Delta t = \Delta t_t$$

$$W = W_{s_n} - \dot{W}_s \frac{\Delta t}{2}$$

The subscript "n" on W_s refers to the aircraft standard weight at the top of the previous or "nth" data interval. This is found by subtracting the fuel used during each of the first "n" data intervals from the aircraft standard weight at the start of the climb, W_{s_0}. For example, if the data being analyzed represents the sixth data pair, the average weight for this interval would initially be estimated to be the standard weight found at the top of the fifth data interval, less the initial estimate of the fuel used during half the sixth inverval.

Start of iterative loop

$$C_L = \frac{W \cos \gamma}{1481\, M^2 S\, \delta}$$

C_D: From drag curve using C_L and M

$$D = 1481\, M^2 S\, \delta\, C_D$$

$$\Delta F_n = F_{n_s} - F_{n_t}$$

$$\Delta D = D - D_t \tag{12.85}$$

$$F_{ex} = F_{ex_t} + \Delta F_n - \Delta D$$

$$\dot{H} = V_{T_s} \frac{F_{ex}}{W} - \frac{\Delta V_{T_s}}{g \Delta t}$$

where ΔV_{T_s} is the desired climb schedule, assumed to be the same as the schedule flown on the test day. Therefore

$$\Delta V_{T_s} = V_{T_{s_2}} - V_{T_{s_1}}$$

where $V_{T_{s_1}}$ and $V_{T_{s_2}}$ are calculated using M_1, M_2 and the standard day temperature at H_{c_1} and H_{c_2}.

$$\gamma = \sin^{-1}\left(\frac{\dot{H}}{V_{T_s}}\right)$$

$$\Delta t = \frac{H_{c_2} - H_{c_1}}{\dot{H}}$$

$$W = W_{s_n} + \dot{W}_s \frac{\Delta t}{2}$$

Check for convergence by comparing the rate of climb, $\dot{H}$, calculated on two successive passes through the above calculation. If the rate of climb has not converged, the calculation beginning with the evaluation of C_L should be

repeated, but now using the updated values for γ, W, and Δt. The following calculations should be performed after convergence has been achieved.

$$C_{L_s} = \frac{W \cos \gamma}{1481\ M^2 S\ \delta}$$

C_{D_s}: From drag curve using C_{L_s} and M

$$D_s = 1481\ M^2 S\ \delta\ C_{D_s}$$

$$\Delta F_n = F_{n_s} - F_{n_t}$$

$$\Delta D = D_s - D_t \qquad (12.86)$$

$$F_{ex_s} = F_{ex_t} + \Delta F - \Delta D$$

$$\dot{H}_s = \frac{V_{T_s} F_{ex_s}}{W} - \frac{\Delta V_{T_s}}{g \Delta t}$$

$$\Delta t_s = \frac{H_{c_2} - H_{c_1}}{\dot{H}_s}$$

$$W_s = W_{s_n} + \dot{W}_s \frac{\Delta t_s}{2}$$

$$P_{s_s} = F_{ex_s} \cdot \frac{V_{T_s}}{W_s}$$

g. Calculate cumulative horizontal distance traveled, time to climb, fuel used, and aircraft weight at the altitude corresponding to the top of the data interval, that is H_{c_2}.

$$\mathrm{Dist}_{n+1} = \mathrm{Dist}_n + V_{T_s} \, \Delta t_s$$

$$\mathrm{Time}_{n+1} = \mathrm{Time}_n + \Delta t_s$$

$$\mathrm{Fuel\ Used}_{n+1} = \mathrm{Fuel\ used}_n + \dot{W}_s \, \Delta t_s \qquad (12.87)$$

$$W_{s_{n+1}} = W_{s_n} - \dot{W}_s \, \Delta t_s$$

The quantities Dist_0, Time_0, and Fuel Used_0 represent the distance, time, and fuel used at the start of the climb. They can be initialized at zero or to other values to account for the distance, time, and fuel used during the takeoff, acceleration to climb speed, and the climb to the initial data altitude. Since they do not enter into the data reduction calculations, the only effect of these initial values is to shift the curves. The standard weight at the initial altitude, W_{s_0}, is an important factor in the data reduction and will change the shape and magnitude of the rate of climb.

After all the data intervals have been analyzed and cumulative values of distance, time, fuel, and weight have been generated, a series of plots can be made as shown in Figure 12.10. The dotted segments represent extrapolated values to sea level. Additional plots of fuel flow and airspeed or Mach could be shown in a similar fashion.

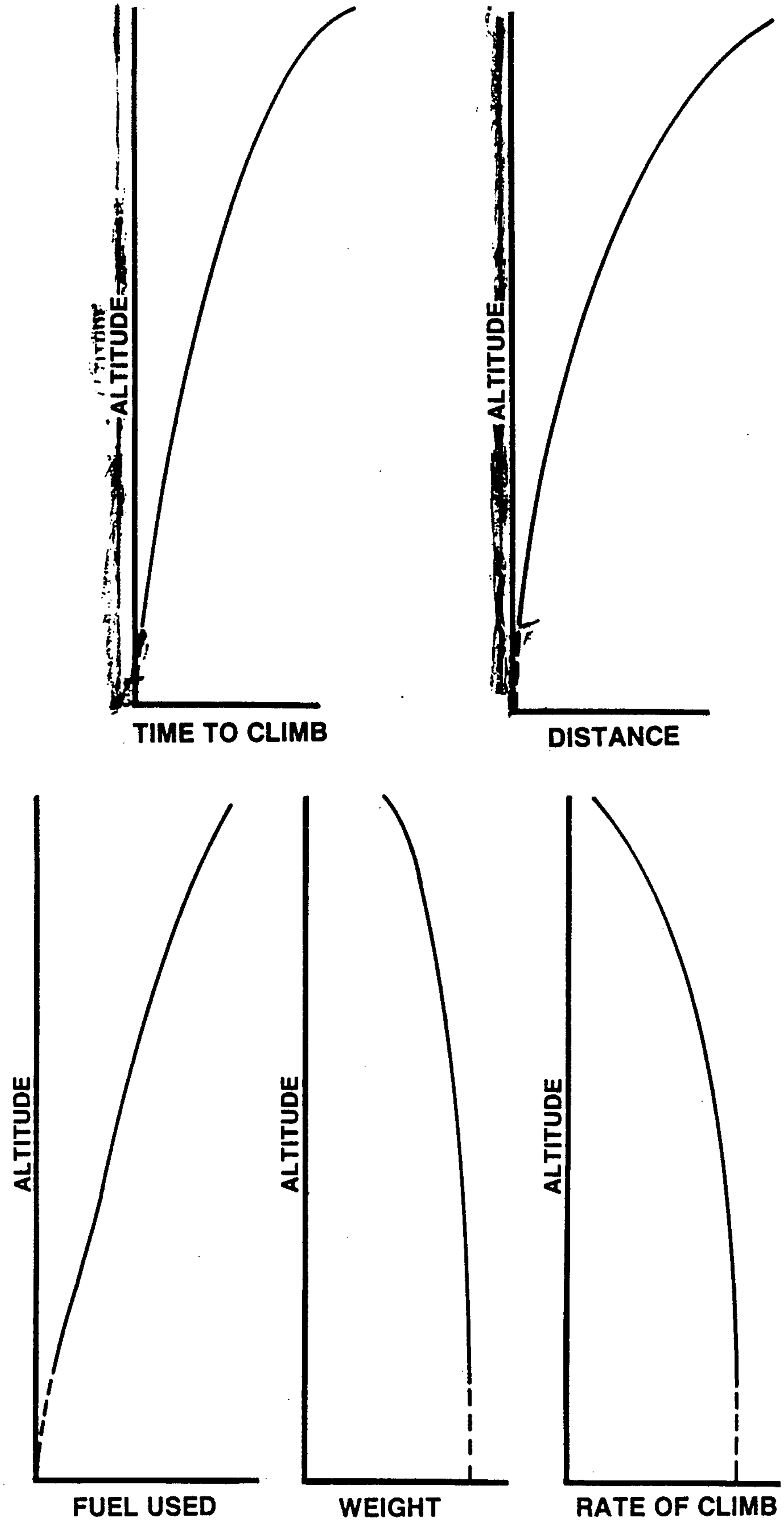

FIGURE 12.10. CLIMB PERFORMANCE SUMMARY

12.5.9 Turn Performance Data Reduction

The turn test is performed to determine an aircraft's sustained turn capability, and associated turn rate and turn radius, as a function of Mach and altitude at a specific power setting. The following data should be recorded during a stable turn.

Normal acceleration, n_t (g's) or time to complete a 360° turn, Δt (sec)

Indicated Altitude, H_i (ft)

Indicated Airspeed, V_i (ft/sec)

Outside Air Temperature, T_a (°K)

Engine RPM, N (RPM)

Aircraft Weight, W_t (lbs)

The altitude at which the data is to be standardized and the corresponding standard weight, W_s, are also required.

a. Use pitot-static relationships to calculate

H_c, M, V_{T_t}, δ_t, θ_t from H_i, V_i, and T_a

V_{T_s}, δ_s, θ_s from M and standard altitude

b. If timed turn, calculate aircraft normal acceleration (load factor).

$$n_t = \sqrt{\left(\frac{2\pi}{\Delta t}\frac{V_{T_t}}{g}\right)^2 + 1} \qquad (12.88)$$

c. Calculate test day parameters

$$C_{L_t} = \frac{n_t W_t}{1481\ M^2 S\ \delta_t}$$

C_{D_t} : From drag curve using C_{L_t} and M (12.89)

F_{n_t} : From thrust curve using M, N, θ_t, δ_t

d. Calculate standard day parameters

F_{n_s} : From thrust curve using M, N, θ_s, δ_s

Predict the aircraft's standard day sustained turn capability by assuming the test day lift can be increased (or decreased) until the increased (or decreased) drag balances the calculated net thrust change between the test and standard day.

$$C_{D_{lim_s}} = C_{D_t} \frac{\delta_t}{\delta_s} + \frac{F_{n_s} - F_{n_t}}{1481\ M^2 S\ \delta_s}$$

$C_{L_{lim_s}}$: From drag curve using $C_{D_{lim_s}}$ and M

$$n_s = \frac{1481\ M^2 S\ \delta_s\ C_{L_{lim_s}}}{W_s} \tag{12.90}$$

$$R_s = \frac{V_{T_s}^2}{g\sqrt{n_s^2 - 1}} \quad \text{for } n_s \geqq 1$$

$$\omega_s = \frac{V_{T_s}}{R_s}$$

The standard day normal acceleration or load factor, n_s (g's), turn radius, R_s (ft), and turn rate, ω_s (rad/sec) should be plotted as a function of Mach, M, for the specific standard altitude and power setting.

12.6 CRUISE PERFORMANCE DATA REDUCTION

The importance of cruise performance should not be understated. Accurate determination of the endurance and range, as well as the corresponding optimum airspeed/altitude profile, is critical in the development and testing of new aircraft. The weight-pressure ratio (W/δ) data collection and reduction technique for cruise performance is described in this section. This method is normally used for turbojet aircraft cruise tests while a constant altitude method is normally used to determine cruise data for a reciprocating engine aircraft. Both are based upon the stable speed-power flight test technique described in Chapter 11. By recording fuel flow data for a sufficiently long stable point at numerous airspeed, altitude, and weight combinations, an estimate of the range and endurance of the aircraft can be obtained. In addition, assuming an accurate model of the thrust characteristics of the engines exist, or can be measured, the speed power flight test technique can be used to estimate the aircraft's drag polar. Finally, ferry range missions can be flown to confirm and refine the range estimates obtained from the speed-power tests.

Test techniques pertaining to the constant W/δ speed-power test are written primarily for the single spool compressor, constant geometry engine. However, they apply equally well to twin spool compressor, variable geometry engines. The data reduction outline, on the other hand, applies only to fixed geometry engines. The power parameters used in the outline are engine speed for single spool compressor engines or the engine pressure ratio, $P_{T_{10}}/P_{T_2}$.

The outlines described here should be modified for more complex engines. Because of the variety of configurations that exist, it is not practical, nor possible, to describe methods for correcting engine data to standard conditions which are suitable for all types. Frequently, it is not immediately evident as to which dimensional analysis methods are applicable. The characteristics of each of the more complex engines should be studied so that methods may be modified for the individual case. Engine manufacturer's charts are a good source of data when making this analysis.

12.6.1 Speed-Power Test, Constant W/δ Method

The constant W/δ method is used to determine the standard day level flight performance of the turbojet aircraft. It is based upon the following mathematical relationships, as were described in Chapter 11.

$$\frac{D}{\delta} = f_1\left(\frac{W}{\delta}, M\right) \tag{12.91}$$

From Buckingham π analysis for jet engines (single spool),

$$\frac{F_n}{\delta} = f_2\left(\frac{N}{\sqrt{\theta}}, M\right) \tag{12.92}$$

Then, since $D = F_n$

$$f_1\left(\frac{W}{\delta}, M\right) = f_2\left(\frac{N}{\sqrt{\theta}}, M\right)$$

$$M = f_3\left(\frac{W}{\delta}, \frac{N}{\sqrt{\theta}}\right)$$

or

$$\frac{N}{\sqrt{\theta}} = f_4\left(\frac{W}{\delta}, M\right) \tag{12.93}$$

The test program covers the range of airspeeds for specific values of $N/\sqrt{\theta}$ and W/δ, and presents the relationship between these parameters and Mach. This gives the relationship between true airspeed, engine speed, and altitude at standard weight and temperature.

In general, the test consists of stabilizing at different airspeeds and power settings while maintaining a constant W/δ. There may be some difficulty in obtaining good data near and below the speed for minimum drag, i.e. the backside of the power curve. While this data will be for speeds below that for maximum endurance, the data is still important. It will be this low speed data that generates the down-turn in the specific range plot (and up-turn in the fuel flow) and therefore defines the peak of the curves. To improve the quality of the data, it is acceptable to allow a slight descent or climb (about 100 ft/min) to maintain airspeed. This method usually gives more reliable data because hysteresis or elastic lag effects in the altimeter are almost eliminated.

For the best results the W/δ should be maintained as close as possible to the desired value, however $\pm$ 2% is usually satisfactory.

12.6.1.1 <u>Preflight Preparation</u>. In order to fly at a constant W/δ certain preflight preparations must be made. The pilot must have charts relating fuel counter to altimeter reading for a constant W/δ. Consider both altitude position error and instrument error when preparing a suitable flight data card. This test is well suited to hand record data.

The following data are required before the necessary charts and tables are prepared:

(1) The empty weight of the aircraft

(2) Fuel density and fuel loading

(3) Altimeter calibrations relevant to altitude and airspeed of the test

(4) Airspeed calibrations (Position and instrument errors)

The following procedures may be used to obtain the charts required to perform the test:

(1) Determine the standard pressure altitude (H_{c_s}) at which the test is to be flown and obtain the corresponding δ from standard atmospheric tables

(2) Determine the standard weight (W_s) corresponding to this altitude

(3) Calculate $(W/\delta)_s$

(4) Obtain the values for the following table:

Altitude	Pressure Ratio	Weight for Standard W/δ
$H_{c_s} + 2000'$	δ_1	W_1
$H_{c_s} + 1000'$	δ_2	W_2
H_{c_s}	δ_s	W_s
$H_{c_s} - 1000'$	δ_3	W_3
$H_{c_s} - 2000'$	δ_4	W_4

Example:

$$W_1 = (W/\delta)_s \times \delta_1$$

(5) Construct a plot of H_c versus weight (Figure 12.11). Given a weight, this plot can be used to determine altitude to fly to achieve the desired W/δ.

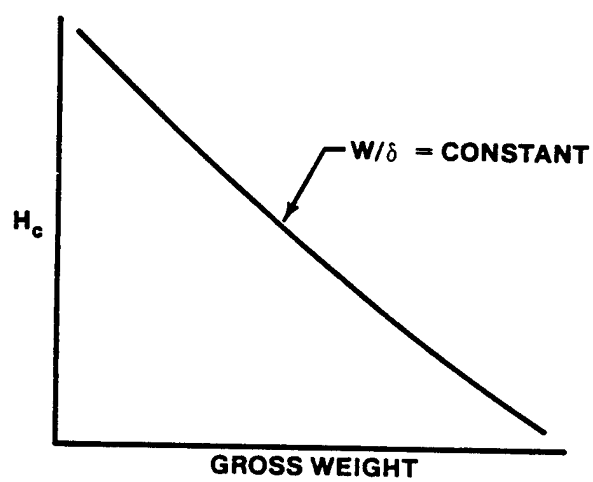

FIGURE 12.11. H_c AS A FUNCTION OF GROSS WEIGHT

(6) Convert gross weight into fuel used during the mission in gallons. Plot H_c versus fuel used as shown in Figure 12.12. The dashed lines shown are the $\pm$ 2% W/δ variation that is permitted. If fuel temperature changes throughout the flight, use an average value for determining fuel density.

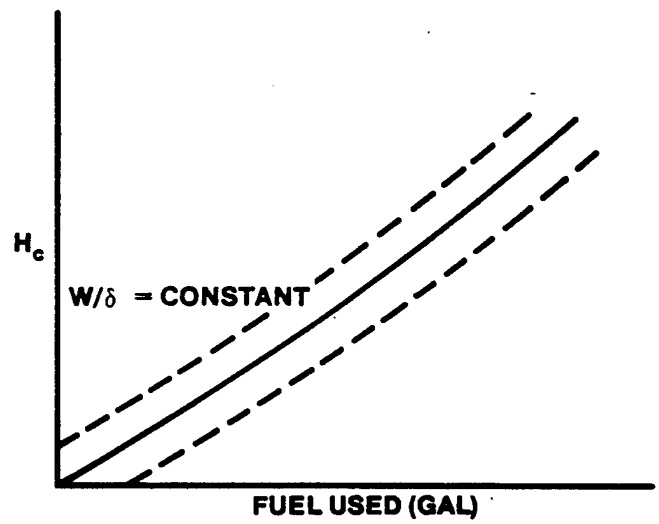

FIGURE 12.12. H_C AS A FUNCTION OF FUEL USED

Figure 12.12 yields the correct altitude to fly at any value of fuel used for a given W/δ.

Note that a plot of fuel counter (F/C) reading versus gross weight is a straight line and is dependent upon the basic weight of the aircraft. If the basic weight of the aircraft changes or the test is flown in another aircraft, this curve (F/C reading versus gross weight) can be easily changed.

(7) Since the values read from the above chart are "altitudes to fly," curves of ΔH_{pc} versus V_{ic} and ΔH_{ic} versus H_i should be used. Give particular attention to the sign (sense) of this correction because the above procedure necessitates going from calibrated values to indicated values.

(8) Below is a suggested data card to be used by the pilot, showing typical entries:

AIM V_i KTS	ACTUAL V_i KTS	ALT FT	F/C START GAL'S USED	F/C END GAL'S USED	T SEC	T_i °C	N_i %	$\dot{W}_f$ lb/hr
V_{max}	456.5	18540	290.1	295.7	1:06.2	20.0	99.9	2000
410	411.0	18810	326.2	331.1	1:14.5	16.5	96.4	1900
370	368.5	19060	352.8	359.3	1:25.3	12.0	93.7	1800

12.6.1.2 <u>In-flight Techniques</u>. The following is a recommended procedure for performing the speed-power test using the constant W/δ method so that flight time may be used efficiently:

(1) Before engine start the pilot should assure himself of the correct fuel loading and that the fuel counters are set correctly.

(2) When approaching within 2000 to 3000 feet of the test altitude, read the fuel counters and extrapolate to account for the 3-5 minutes it will take to stabilize on condition. Obtain an "altitude to fly" for the first data point based upon this fuel counter estimate.

(3) Using the aim airspeed from the flight data card, enter the ΔH versus V_i curve and determine the correction to apply to H_c. Using the allowed fuel, establish a stable point at the aim airspeed and corrected altitude.

(4) Record the fuel counter reading and start the stop watch when the aircraft is stabilized within 2 percent of the desired W/δ. Fly the aircraft at the required altitude for a minimum of one minute, then record the fuel counter reading and other data. Ideally, the fuel counter reading for the correct W/δ would occur midway through the timed period. For low fuel rates a longer stable point may be required. This is especially true for an instrumentation system with a fuel flow resolution of 0.1 gallons or more, such as used at the TPS. In this case, a minimum two minute stable point is required to obtain accurate estimates of fuel flow.

The pilot should be absolutely certain that the aircraft is stabilized before recording data. If the airspeed changes more than $\pm$ 2 knots using the front side technique or the vertical velocity exceeds $\pm$ 100 ft/min using the back side technique, the point should be repeated.

(5) Obtain enough stabilized points to completely define the fuel flow versus velocity curve at the particular W/δ tested. By plotting fuel flow versus velocity during the mission, the pilot can be assured he has taken a sufficient number of points.

(6) The pilot can expedite stabilizing the aircraft by proper trimming, pitch control by outside reference, and recording data in an organized sequence. The aircraft should be trimmed for hands-off flight when stabilized. Altitude control on the front side points and airspeed control on back side point can be controlled precisely by the attitude method. It will be found that the majority of the data may be recorded while waiting for the aircraft to stabilize.

12.6.1.3 W/δ Data Reduction. Pairs of data points, representing the start and stop of each stable point, should be recorded for use in the data reduction equations listed below. The following parameters should be recorded.

Altitude, H_i (ft)

Indicated Airspeed, V_i (ft/sec)

Engine RPM, N (RPM)

or

Engine Pressure Ratio, $P_{T_{10}}/P_{T_2}$ (EPR)

Outside Air Temperature, T_a (^{o}K)

Aircraft Weight, W_t (lb)

Time, t (sec)

The standard altitude, H_{c_s} (ft), and corresponding standard weight, W_s (lbs) are also required.

The following steps should then be performed

a) Use pitot-static relationships to calculate:

$$H_{c_1}, M_1, T_{a_1}$$

$$H_{c_2},\ M_2,\ T_{a_2}$$

where the subscripts 1 and 2 refer to the start and stop times respectively.

b) Calculate test day average values:

$$H_c = \frac{H_{c_1} + H_{c_2}}{2}$$

$$M = \frac{M_1 + M_2}{2}$$

$$T_t = \frac{T_{a_1} + T_{a_2}}{2}$$

$$W_t = \frac{W_{t_1} + W_{t_2}}{2}$$

$$N = \frac{N_{t_1} + N_{t_2}}{2}$$

(c) Use pitot-static relationships to calculate the following test and standard day parameters.

V_c, δ_t, θ_t calculated from H_c, M, and T_t

δ_s, θ_s calculated from H_{c_s}

(d) Calculate average fuel flow rate:

$$\dot{W}_{f_t} \cong \frac{W_{t_1} - W_{t_2}}{\Delta t}$$

where Δt is the duration of the stable point.

(e) Calculate range performance parameters.

The following equations predict the range performance on a standard day. As with all data reduction, it is necessary to determine which parameters are invariant between test and standard day conditions. As stated in Chapter 11, aircraft performance is fully described by two independent variables, Mach, and weight to pressure ratio, W/δ. Corrected fuel flow, $\dot{W}_f/\delta\sqrt{\theta}$, corrected RPM, $N/\sqrt{\theta}$, and range factor, $V_t W/\dot{W}_f$, are each a function of only these two variables and therefore are invariant between test and standard day. This means, for example, the corrected fuel flow measured on the test day will be precisely the same value as would have been measured on a standard day at the same Mach and W/δ. This is true even if the test day weight is different than the standard weight, provided the test day altitude is such that it corresponds to the same value of W/δ. On the other hand, other parameters such as fuel flow, $\dot{W}_f$, and specific range, SR, are a function of more than two variables, in particular M, W/δ, and altitude. They are therefore not invariant between test and standard day. This fact is shown below in the plot of SR versus Mach.

The following parameters should be calculated:

(1) Compare test and standard day weight to pressure ratio

$$\%\ \text{Error} = \frac{W_t/\delta_t - W_s/\delta_s}{W_s/\delta_s} \times 100 \qquad (12.94)$$

If the % Error is greater than 2%, then this data point should be ignored since no equations are given to correct for W/δ errors.

(2) Calculate standard day specific range

$$\dot{W}_{f_s} = \dot{W}_{f_t} \frac{\delta_s\sqrt{\theta_s}}{\delta_t\sqrt{\theta_t}} \qquad (12.95)$$

$$SR_s = \frac{Ma_0\sqrt{\theta_s}}{\dot{W}_{f_s}} \qquad (12.96)$$

where a_0 is the standard day speed of sound at sea level. A set of specific ranges, SR_s, for various Mach, at one desired weight to pressure ratio, W/δ_s, should be plotted as shown in Figure 12.13. Because specific range is a function of more than two variables, it

is valid only for the Mach, weight-pressure ratio, and standard altitude for which it has been calculated. This is indicated on the plot shown below, where H_1 is the standard altitude corresponding to the desired W/δ.

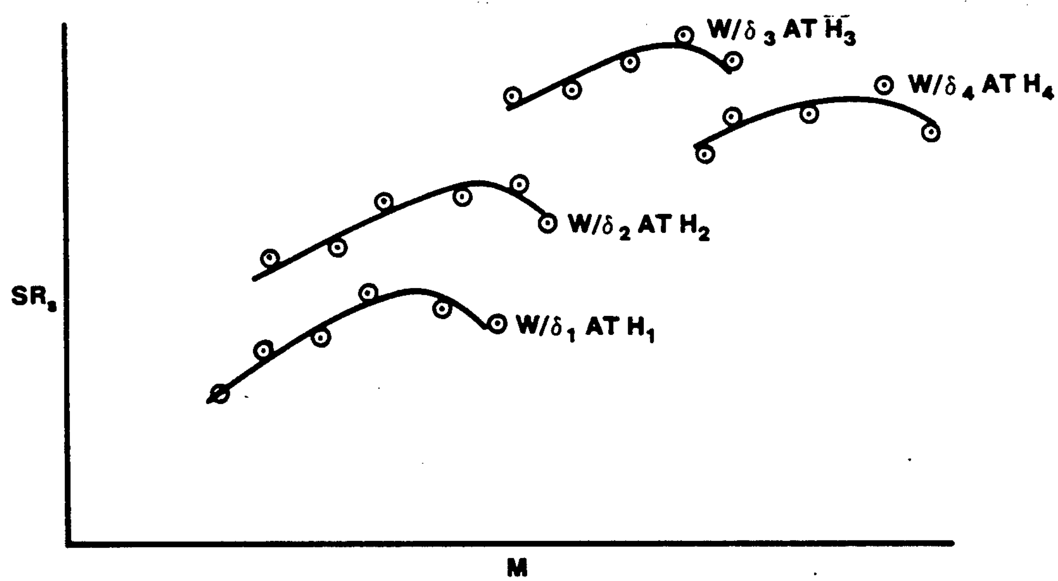

FIGURE 12.13. SPECIFIC RANGE VERSUS MACH FOR VARIOUS WEIGHT-PRESSURE RATIOS

The data plotted above can now be used to determine the optimum Mach and weight to pressure ratio for maximum range. The range factor, RF, given by

$$RF = (W_s)\ SR_s \tag{12.97}$$

should be calculated for each of the four points (or as many as there are W/δ curves) representing the maximum of each specific range curve. The standard weight, W_s, in this equation corresponds to each standard altitude. For example, for the curve corresponding to W/δ_2 at H_2, the standard weight is given by

$$W_{s_2} = \left(\frac{W}{\delta}\right)_2 \delta_2 \quad \text{where } \delta_2 \text{ is found from } H_2 \qquad (12.98)$$

These four range factors, as well as the corresponding Mach, should be plotted versus W/δ as shown in Figure 12.14.

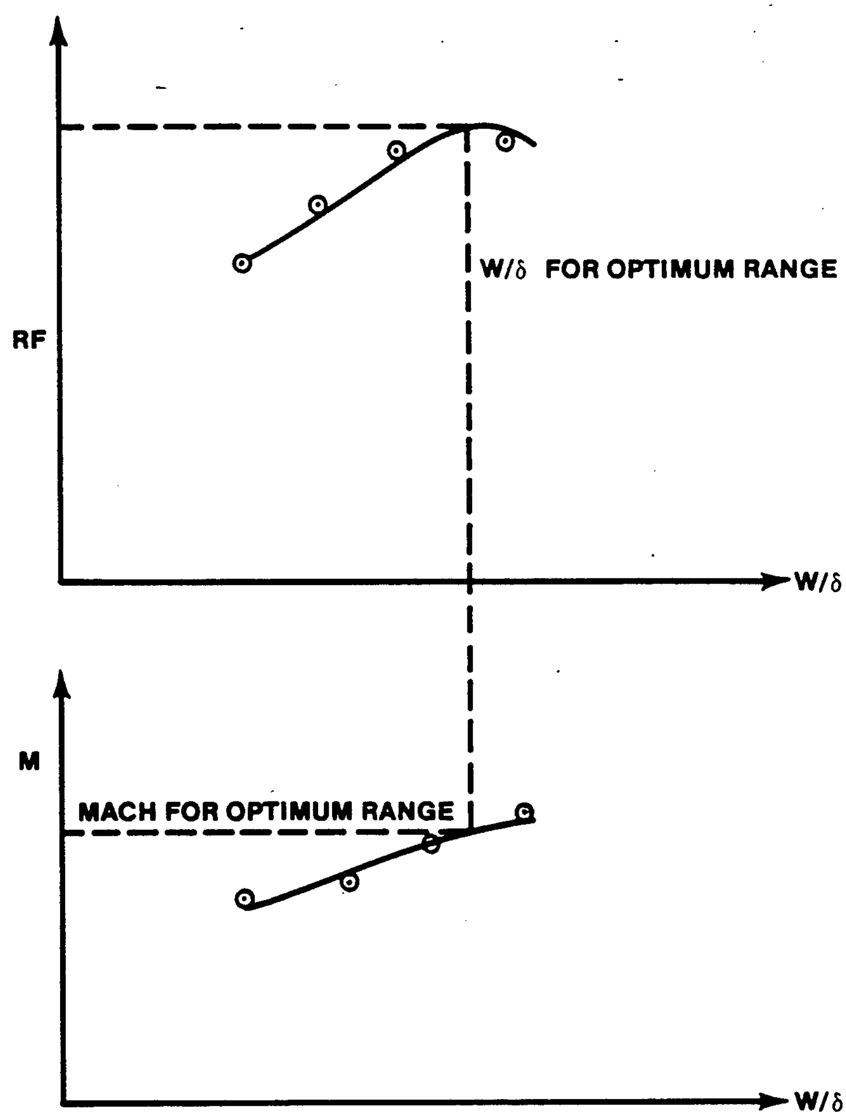

FIGURE 12.14. RANGE FACTOR AND MACH VERSUS WEIGHT-PRESSURE RATIO

The range factor corresponding to the optimum Mach and weight to pressure ratio can now be determined. From theory, the range for this optimum range factor can be calculated from

$$R = RF \ln \frac{W_i}{W_f} \tag{12.99}$$

where W_i is the aircraft weight at the start of the cruise and W_f is the weight at the end of the cruise. An aircraft's maximum ferry range can be calculated by using the maximum weight for W_i, and the minimum weight for W_f. Maximum W_i should take into account the fuel used for ground operations, takeoff, acceleration to climb speed, and climb to the altitude for the start of the cruise climb. Minimum W_f is found using MIL-C-5011A fuel reserve. The distance traveled during the climb and descent should be added to the range during the cruise climb to determine the total ferry range.

12.6.1.4 Drag Polar Determination. The aircraft's drag polar can be predicted from the speed-power test, provided an accurate model of the thrust characteristics of the engine exists or can be measured. The calculations necessary to plot the drag polar are listed below.

a) Calculate test day lift coefficient and drag coefficient

$$C_{L_t} = \frac{W_t}{1481\ M^2 S\ \delta_t}$$

$$F_{n_t} = \text{From thrust curve using M, N, } \theta_t \text{ and } \delta_t$$

Since the aircraft is unaccelerating during the speed-power test, thrust equals drag.

$$D_t = F_{n_t}$$

Test day drag coefficient is calculated from

$$C_{D_t} = \frac{D_t}{1481\ M^2 S\ \delta_t}$$

The drag polar can be plotted as shown in Figure 12.15. Since the drag polar is a function of Mach, only those points of equal Mach can be connected with a curve. However, for all practical purposes, the drag polar is independent of Mach below M = 0.75. Hopefully enough data will be below this Mach to permit an accurate curve of the subsonic drag polar and the corresponding Oswald efficiency factor "e" to be generated. Test points at higher Mach can be used to estimate the effect of Mach on the parasitic drag.

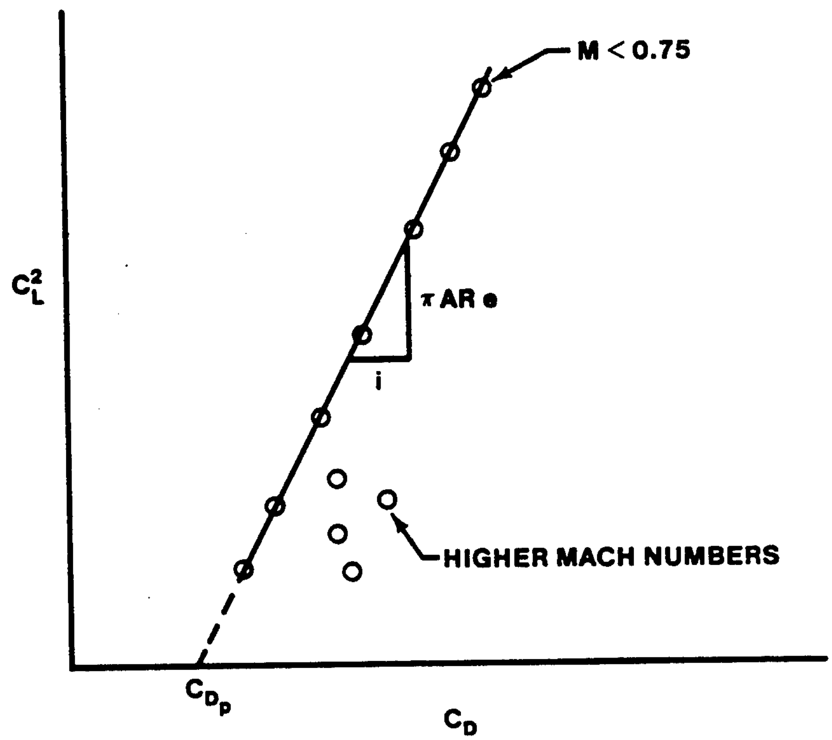

FIGURE 12.15. DRAG POLAR

12.6.2 Range Cruise Control Test

The range cruise control test (Ferry Range Mission) is used to verify and refine the estimates of the range performance generated during the W/δ tests. Specifically, it should be used to check the optimum W/δ and the total ferry range. Usually a series of flights will be flown at W/δ above and below the

predicted optimum W/δ. The standard day range from each of these can be used to determine the actual optimum W/δ and compared with the data predicted by W/δ testing.

Planning of the time or distance flown during the cruise portion of the test is outlined below:

		Fuel Used	Fuel Remaining
1.	Prior to Eng Start		
2.	Engine Start + Taxi		
3.	Takeoff and Accelerate to Climb Schedule		
4.	Climb		
5.	Cruise		
6.	Fuel Reserve		

Estimated fuel used for engine start, taxi, takeoff, and acceleration to climb schedule is obtained from manufacturer's charts. Fuel used in the climb is obtained from the check climb test. Fuel reserve is determined by reference to MIL-C-5011A. The total of these fuel increments subtracted from the total fuel available gives the amount of fuel available for the cruise portion of the test.

The following is a suggested flight data card to be used by the pilot:

Pilot ______	A/C No. ______	Date ______

W/δ = ______

Data Point	Time	F/C	V_i	H_i	T_i	RPM	$\dot{W}_f$
Prior Eng Start							
Start							
Taxi							
T. O.							
Start of Climb							
End of Climb							
Start Cruise							

The pilot should plan the fuel used during the climb as a function of the W/δ profile. When on the cruise schedule, the crew should record data often enough to obtain at least 10 points.

A fuel counter versus altitude chart for the desired W/δ can be prepared as is outlined in the preflight section of the speed-power at constant W/δ test.

12.6.2.1 Inflight Techniques. A recommended procedure for performing the range cruise control test is:

(1) Prior to engine start, check that the correct amount of fuel is onboard and that the fuel counter is set correctly.

(2) Record data at each point planned, i.e., engine start, taxi, takeoff, start of climb and end of climb.

(3) Upon reaching the altitude that corresponds to the fuel counter reading for the desired W/δ, set up the cruise climb at the desired Mach.

(4) Increase altitude as the fuel counter is decreased to maintain a constant W/δ by performing a shallow climb. An alternate method is to hold a constant altitude and stair-step the aircraft in increments of 100 to 200 feet. Cruise should begin in the stratosphere and the Mach and RPM should remain constant (using a constant velocity for the Mach is preferred due to the accuracy of the instruments). If cruise should begin below the tropopause, a slight decrease in RPM will be required initially. Equation 12.93

shows that for a given W/δ and Mach a constant $N/\sqrt{\theta}$ is required and therefore engine RPM is decreased as T_a decreases to hold $N/\sqrt{\theta}$ constant.

Throttle movements should be held to a minimum. If turns are required, they should be very shallow.

12.6.2.2 <u>Ferry Range Data Reduction</u>. Record the following parameters throughout the ferry range test. Collect sufficient data points to minimize the effect of errors in reading the data.

Altitude, H_i (ft)

Indicated Airspeed, V_i (ft/sec)

Outside Air Temperature, T_a ($^{\circ}$K)

Aircraft Weight, W_t (lb)

Time, t (sec)

The data reduction outline presented below also requires the standard day initial cruise weight, W_{s_i} (lb), and the final cruise weight, W_{s_f}, (lb). Pitot-static relationships should be used to calculate the aircraft's true airspeed, V_T, (ft/sec) and Mach. The test day W/δ for each set of data points should also be calculated to ensure the test pilot flew the planned Mach and W/δ. The test day total range (air miles) is found by numerically integrating the true airspeed with respect to time. That is

$$R_t = \sum_{j=1}^{n} V_j \Delta t_j \tag{12.100}$$

where Δt_j represents each of the "n" time intervals between data points and V_j is the average true airspeed during that time interval. The test day average range factor, RF_t, is found from

$$RF_t = \frac{t}{\ln\left(\frac{W_i}{W_f}\right)} \tag{12.99}$$

where W_i and W_f are the test day initial and final cruise weights respectively. Standard day cruise range is then predicted using

$$R_s = RF_t \ln\left(\frac{W_{s_i}}{W_{s_f}}\right) \tag{12.101}$$

The total range capability of the aircraft can now be evaluated. Total range is equal to the sum of nautical air miles traveled during climb plus nautical air miles traveled during cruise. Range credit is not allowed for takeoff and acceleration to climb speed. Distance traveled during the climb is obtained from the check climb test and distance traveled during cruise is computed by using the range factor just determined in Equation 12.101 where W_{s_i} is the standard weight at the start of cruise (end of climb) using fuel allowances for ground time and acceleration of climb speed and fuel for climb from the check climb test. W_{s_f} is the standard weight at the MIL-C-5011A fuel reserve requirements. Figure 12.16 shows typical format for ferry range data.

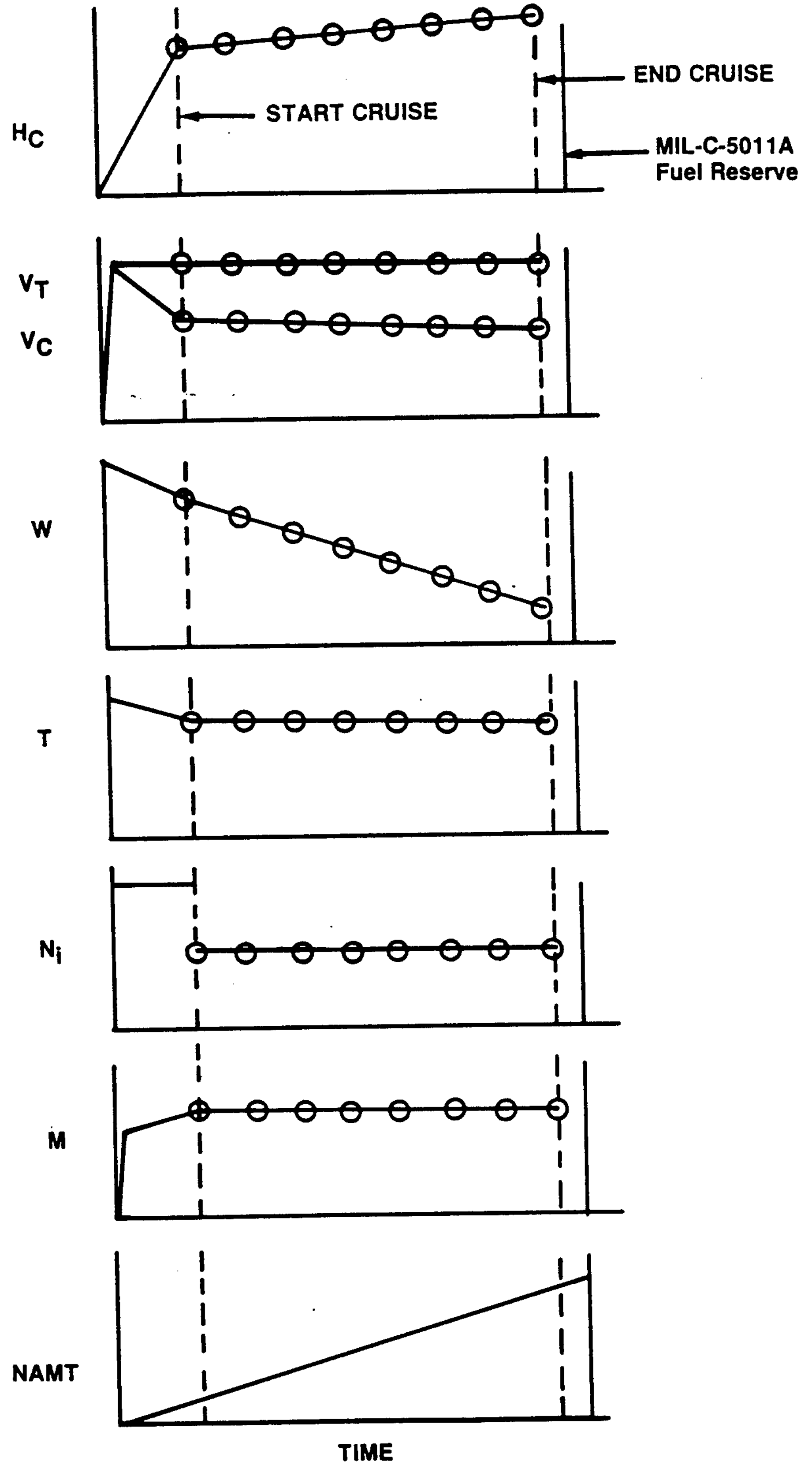

FIGURE 12.16. FERRY RANGE